(12) LEVEL II

AD-E300 368

DNA 4251F

(6)

SKIN FRICTION DRAG DUE TO NUCLEAR DAMAGE.

The Boeing Aerospace Company
P.O. Box 3999
Seattle, Washington 98124

(10) E. /York
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(11) 30 September ~~19~~77

(12) 154p.

(9) Final Report, ~~for Period~~ Febr~~uary 1977~~—Sept~~ember~~ 1977,

CONTRACT No. DNA 001-77-C-0090

(15)

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1. REPORT NUMBER: DNA 4251F	2. GOVT ACCESSION NO.
3. RECIPIENT'S CATALOG NUMBER	
4. TITLE (and Subtitle): SKIN FRICTION DRAG DUE TO NUCLEAR DAMAGE	5. TYPE OF REPORT & PERIOD COVERED: Final Report for Period February 1977—September 1977
	6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s): E. York, W. Fleener	8. CONTRACT OR GRANT NUMBER(s): DNA 001-77-C-0090
9. PERFORMING ORGANIZATION NAME AND ADDRESS: The Boeing Aerospace Company, P.O. Box 3999, Seattle, Washington 98124	10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: Subtask N99QAXAE506-03
11. CONTROLLING OFFICE NAME AND ADDRESS: Director, Defense Nuclear Agency, Washington, D.C. 20305	12. REPORT DATE: 30 September 1977
	13. NUMBER OF PAGES: 158
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)	15. SECURITY CLASS (of this report): UNCLASSIFIED
	15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

This work sponsored by the Defense Nuclear Agency under RDT&E RMSS Code B342077464 N99QAXAE50603 H2590D.

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Skin Friction
Thermal Damage
Equivalent Sand Roughness
Nuclear Damage

sq cm

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Typical thin aircraft skin samples were fabricated of aluminum and fiberglass and painted with black enamel and other coatings. The samples were exposed to fluence levels from 20 to 200 cals/cm2 to produce simulated nuclear thermal damage. Their skin friction was then measured at Mach numbers of 0.5 and 0.8 and Reynolds' numbers of 6 to 16 million. An equivalent sand roughness was calculated from the drag. Results showed an initial increase in roughness which then remained constant until the skin melted, debonded or suffered other severe damage.

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Summary

Twenty nine samples were fabricated of thin skins and coatings which are typical of aircraft secondary structure. They included aluminum and fiberglass-epoxy of various gages bonded to a honeycomb substrate, with 250°F and 350°F cure bonds. The coatings were mainly black enamel (for heat absorption) with seven other coatings tested on the .030 in. fiberglass-epoxy. Up to twelve test specimens were cut from each sample sheet, and exposed to thermal fluence levels from 20 to 200 cals/cm^2 in the Air Force Materials Laboratory quartz lamp facility.

The specimens were then sorted for representative thermal damage to the surface, and their skin friction was measured at M = 0.5 and 0.8. The results were converted to equivalent sand roughness using Schlicting's correlation, and plots made of roughness vs. heat absorbed.

The results showed that there was a fairly rapid increase in roughness as soon as the paint blistered (10-20 cals/cm^2) and then stayed fairly constant with increasing heat absorption until the skin debonded, melted or suffered other damage. This made the profile drag the dominant effect and skin friction or roughness no longer was meaningful. The aluminum specimens generally showed a lower value of roughness than the fiberglass ones and were probably not in the fully rough regime.

Surface measurements were made with a profile meter, but no consistent correlation could be obtained between the roughness measurements made in this manner and those obtained from the wind tunnel drag measurements.

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TABLE 1-1

Conversion factors for U.S. customary to metric (S.I.) units

To convert from	To	Multiply by
mils	millimeters (mm)	0.0254
inches (in.)	centimeters (cm)	2.54
miles	kilometers	1.6093
kilotons	terojoules	4.183
pounds/in^2	kilopascals	6.89
farenheit	degrees kelvin (K)	(t°f + 459.67)/1.8
cals^2/cm^2	megajoule/m^2	.0418

1.0 INTRODUCTION

An aircraft exposed to a nuclear weapon blast environment may experience damage which would degrade its performance but would still leave the aircraft in flyable condition. Therefore, the degree of damage that an aircraft can experience and complete its mission is of particular interest. The ability of an aircraft to complete its mission will depend on the degradation of performance of various weapon system elements including crew efficiency, structural capability, avionics subsystem performance, propulsion system efficiency, control system efficiency and aerodynamic efficiency (lift/drag ratio). Increased drag may be the result of distinct responses to the nuclear weapon environment: structural deformation due to the overpressure, gust or thermally induced structural loading, and skin roughening due to thermal radiation.

Previous laboratory investigations have shown that typical aircraft surfaces such as painted aluminum skin, fiberglass honeycomb or aluminum honeycomb panels become quite rough when exposed to severe thermal radiation pulses similar to those produced by nuclear weapons. This increase in surface roughness has been recognized as having the potential of increasing the skin friction drag on an aircraft to the degree that the ability of the aircraft to complete its mission will be degraded.

The increase in friction drag can be calculated using analytical methods because the dominant controlling parameters are definable for a broad range of aircraft configurations. This report provides the required basic empirical data for definition of the change in skin friction drag of an aircraft due to a nuclear encounter.

1.1 NUCLEAR THERMAL EFFECTS ON AIRCRAFT SURFACES

The thermal radiation released by a nuclear weapon detonation constitutes a major threat to non-metallic and thin skin metal aircraft structure. This arises partly due to the fact that 35% of the total energy released by a nuclear weapon detonated in the atmosphere is in the form of thermal radiation. Further increasing the thermal radiation threat to aircraft is the fact that skin

structures are made of lightweight thin materials having little capability of absorbing or dissipating thermal energy without degradation of their mechanical strength properties and thus the structural capability for which they were designed. Consequently, the thermal response of an aircraft must be determined and included in every study that assesses the survivability of an aircraft system in a nuclear weapon induced environment.

The thermal response of specific aircraft structures can be determined and classified into various damage levels corresponding to increasing levels of exposure. However, to determine the net effect of thermal exposure on an entire aircraft system, one must consider not only specific component response, but the cumulative effects of all damaged components on the performance of the aircraft. Most thermally susceptible components on aircraft are secondary structure or aerodynamic fairings. Damage or complete failure of these structures will not jeopardize the aircraft's structural integrity. Thermal damage to them, however, may result in complete loss of some skin structures and extensive damage to others, but still retain sufficient performance to complete its assigned mission. Another possibility is that the aircraft may sustain extensive but superficial damage which, while posing no immediate hazard to its crew, will sufficiently degrade aerodynamic performance to preclude completion of the assigned mission. In such a case, the superficial thermal damage will have resulted in an effective "kill". One of the criteria for determining whether or not a damaged aircraft can complete its mission, therefore, must be an assessment of the impact in aerodynamic drag caused by thermal radiation.

1.2 NUCLEAR THREAT

In order to estimate the thermal pulse which should be applied in the test to simulate a nuclear encounter we have been guided by possible overpressure levels. Glasstone gives the following approximate values for overpressure:

	1 Mile	3 Miles	10 Miles
1 KT	1 psi	.2 psi	--
100 KT	10 psi	1.5 psi	.3 psi
1 MT	50 psi	5.0 psi	1.0 psi

* "The effects of nuclear weapons", (Rev.) S. Glasstone Ed. U.S. Atomic Energy Commission 1964.

The corresponding thermal pulses (for 10-50 mile visibility) are:

1 KT	1 cal/cm^2	.1 cal/cm^2	0.005 cal/cm^2
100 KT	100 cal/cm^2	10 cal/cm^2	0.5 cal/cm^2
1 MT	1000 cal/cm^2	100 cal/cm^2	5. cal/cm^2

Thus the test thermal pulses were chosen from the minimum necessary to cause paint damage -- about 20 cal/cm^2 to a maximum of about 100 cal/cm^2, skin damage. Beyond this level would certainly be approaching the sure kill level for aerospace structures.

1.3 FLIGHT CONDITIONS

For a subsonic cruise aircraft, the maximum local Mach number (M) at a cruise of M = .85 is approximately 1.3. However, this supersonic region is over the center third of the wing where the skin structure is thick and not as susceptible to thermal damage as on the leading and trailing edges.

Flight Reynolds numbers corresponding to M = .8 at 30,000 ft. are 2.3×10^6 per foot, or approximately 3×10^7 and 3×10^8 for the wing and aft fuselage respectively. For M = .5 at 3,000 ft. (a base-escape mode) the Reynolds numbers are about doubled. The wind tunnel test range possible for this program was 6×10^6 to 15×10^6.

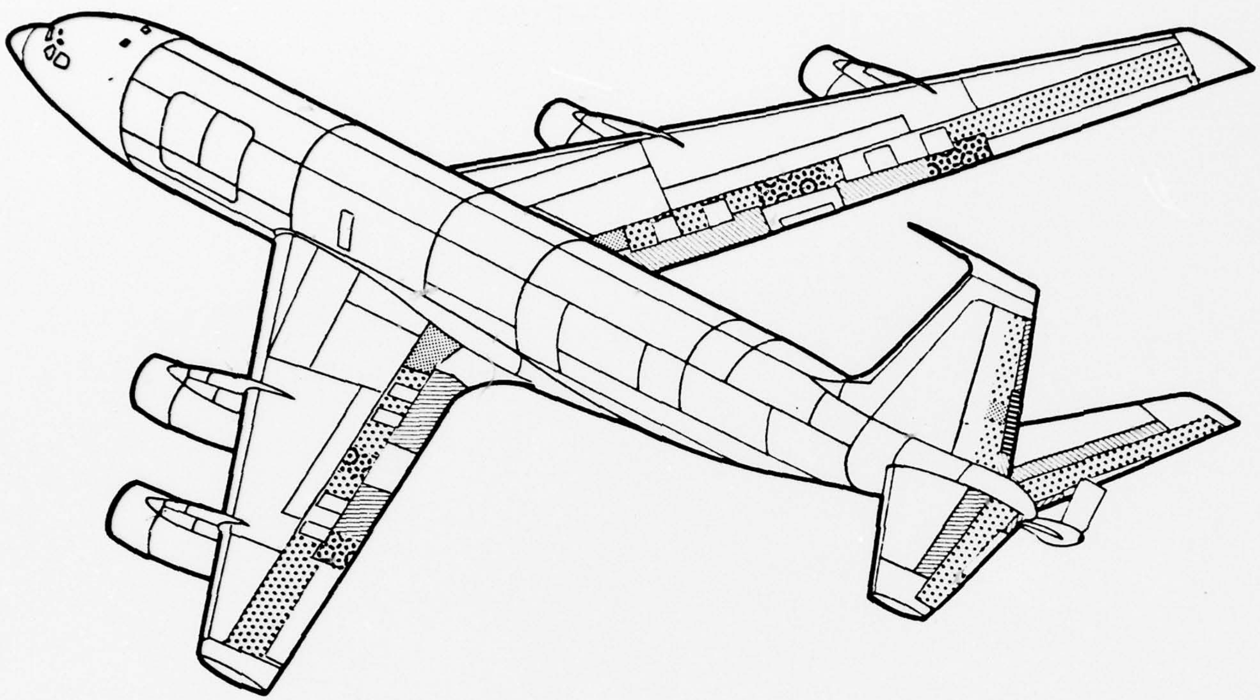
The boundary layer thickness on the aircraft would vary from 2 inches on the wings or stabilizers up to 10 inches on the aft fuselage. In the wind tunnel it was .2 to .4 inches.

2.0 TEST SPECIMEN SELECTION AND FABRICATION

A survey of many operational combat aircraft shows that the thin-skin thermally vulnerable structure which characterizes most aircraft construction can be grouped into three categories. These structures include most fiberglass and aluminum honeycomb sandwich panels and some of the thinner aluminum sheet panels used in semi-monocoque construction. Figures 2-1 and 2-2 show the location of the thin-skin aluminum honeycomb panels on the KC-135A and the thin-skin materials used on the 747. Although the 747 is the only major aircraft that currently uses extensive fiberglass paneling for secondary skin materials, almost all aircraft use fiberglass for radome and antenna fairing construction. The new generation of tactical-aircraft use thin-skin aluminum honeycomb paneling extensively. The most universal use of these thin-skin materials is found on control surfaces, flaps, spoilers and for leading edge and trailing edge skins on the wings and empennage (fore and aft of the primary torque box structure spars). Much heavier gage metals are used on all aircraft between the main spars (for carrying bending and torsional loads) and on pressurized fuselage sections. These heavier skin structures are not of concern from the thermal standpoint since their heat sink capacity is great enough that blast-overpressure rather than thermal, becomes the limiting environment.

2.1 SPECIMEN SELECTION

The thin-skin aircraft materials generally are manufactured in uniform gages. Aluminum honeycomb face sheets are rarely found thinner than 0.012 inch and progress to thicker dimensions in 0.004 inch increments up to about 0.032 inch. Fiberglass skin structures are generally of laminated construction, coming in 0.0045 inch or 0.01 inch per ply dimensions. Minimum fiberglass face skins are rarely found thinner than 0.0135 inch (3 ply x 0.0045 in.) or 0.02 inch (2 ply x 0.01 in.). Most radome construction employs fiberglass outer face sheets of 0.03 inch or more. Aluminum sheet (semi-monocoque construction) is seldom used in gages less than 0.032 inch. Uniform increases in aluminum sheet generally ascend by 0.01 inch increments, i.e., 0.04, 0.05



ALUMINUM HONEYCOMB, 0.016" OUTER SKIN - 225 SQ. FT./8.7 SQ. FT. VERTICAL TAIL

0.018 BONDED ALUMINUM SKIN - 255 SQ. FT./83.9 SQ. FT. VERTICAL TAIL

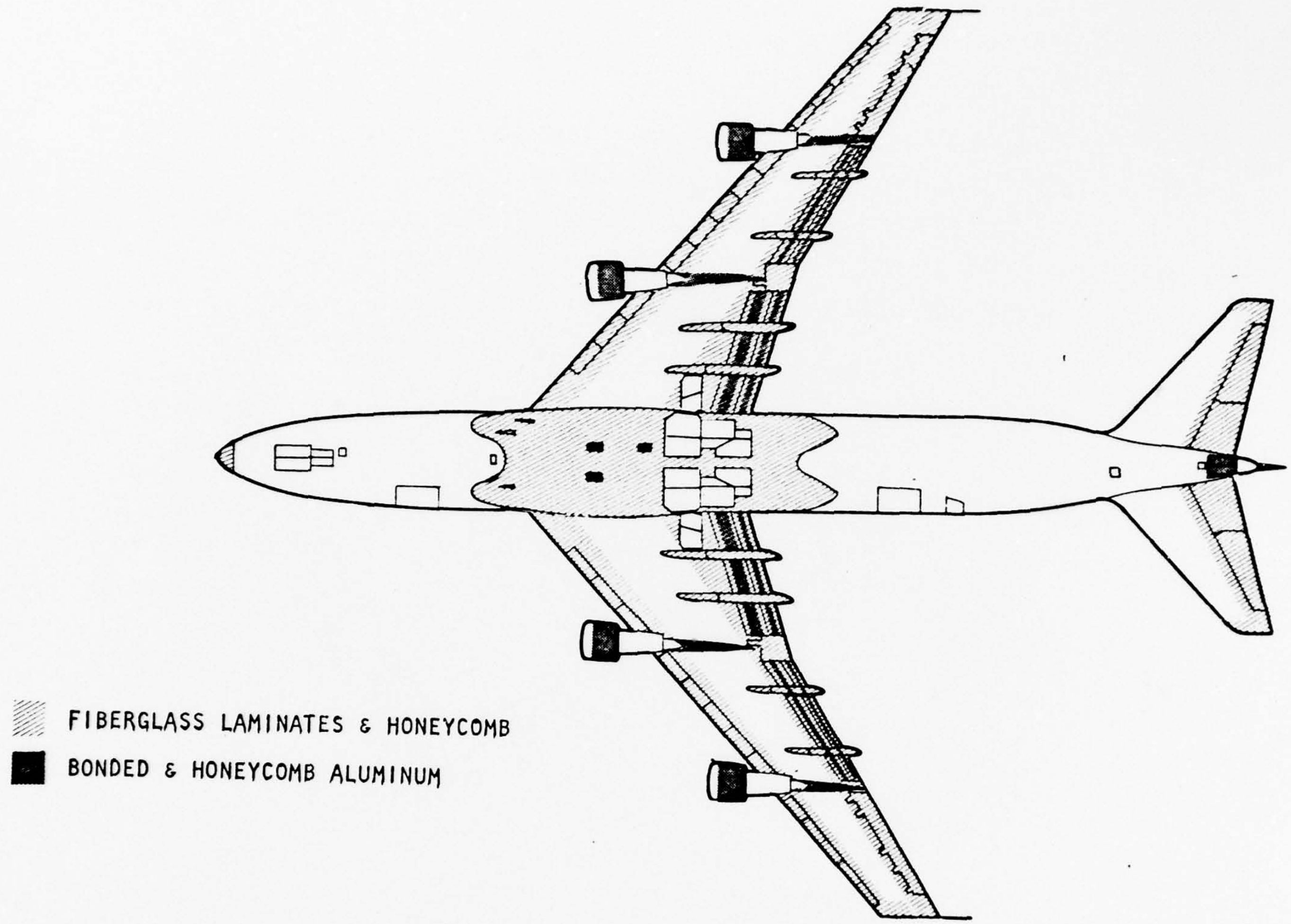
ALUMINUM HONEYCOMB, 0.010" OUTER SKIN - 7.0 SQ. FT.

0.020 BONDED ALUMINUM SKIN,- 33 SQ. FT./3.5 SQ.FT. VERTICAL TAIL

ALUMINUM HONEYCOMB, 0.020" OUTER SKIN - 54.2 SQ.FT

Thin Skin Panels on the KC-135

FIGURE 2-1



Thin Skin Panels on the 747

FIGURE 2-2

inch, etc. Heavier gage aluminum sheet is often composed by the use of bonded or riveted doublers, triplers, etc., to achieve the required structural strength.

Aircraft thin-skin structures most susceptible to thermal damage are listed below:

a. Aluminum honeycomb sandwich with face sheet thicknesses of 0.012, 0.016, 0.020, 0.025 or 0.032 inch nominal.
b. Fiberglass laminate honeycomb sandwich with face sheet thicknesses of 0.0135, 0.020, 0.030 or 0.050 inch nominal.
c. Aluminum sheet 0.04, 0.05 or 0.06 inch nominal thickness.

The thermal response of aluminum is not significantly affected by alloy or heat treat. Therefore, a single alloy/heat treat, typical for aircraft skins, was used for face sheet of all aluminum alloy specimens. Aluminum honeycomb specimens used a high temperature (350^{O}F) cure adhesive for one set (5 face sheet thicknesses) and a low temperature (250^{O}F) cure adhesive for another set.

Two sets of fiberglass specimens were constructed in the 0.0135, 0.020 and 0.030 inch face thicknesses. One set utilized a high temperature (350^{O}F) cure epoxy and the other set used a low temperature (250^{O}F) cure epoxy. Both sets used "E" glass. The 0.050 inch face sheet specimens used "S" glass and a high temperature cure epoxy.

Coatings can significantly affect the thermal response and at least the initial damage mode of specimens. The exterior coating system most commonly used on current military aircraft is a polyurethane enamel, MIL-C-83286, applied over an epoxy primer, MIL-P-23377. This coating system was applied on all configurations of specimens. A black enamel was used to maximize thermal absorption. Additional, widely used coating systems were applied on a single configuration, i.e., 0.030 inch face sheet low temperature cure epoxy-fiberglass laminate honeycomb sandwich. The use of a common substrate helped to better identify differences in thermal damage mode among the various coating systems. The following, additional coating systems were tested:

a. Polyurethane Enamel, MIL-C-83286, over Polysulfide Primer
b. MIL-L-81352 Lacquer over MIL-P-23377 Primer
c. Astrocoat Rain Erosion Resistant Coating System, black over white
d. Astrocoat, all black, MIL-C-83231
e. Astrocoat, non-yellowing white, MIL-C-83445
f. Fluorocarbon Rain Erosion Resistant Coating System, all black
g. Fluorocarbon, white

Table 2-1 gives a listing of the specimens selected.

2.2 TEST SPECIMEN FABRICATION

All specimens were fabricated as honeycomb sandwich with a total thickness of approximately 0.4 inches. The honeycomb sandwich construction is typically used in aircraft panels for all of the test panel materials/thicknesses except the 0.032 inch magnesium and the 0.040, 0.050 and 0.063 aluminum. In the case of these 4 materials the honeycomb sandwich construction was used only for a test program convenience, i.e. to standardize specimen thickness and to support and stiffen the face sheet material.

All honeycomb core was 0.375 inch nominal depth. Aluminum honeycomb core used for specimen series 1-X through 10-X was 5052-H39 aluminum alloy, 3/16 inch cell size, 0.010 inch wall thickness and 3.1 lb/ft^3 density. All 250^{O}F cure epoxy fiberglass and epoxy graphite specimens used Nomex honeycomb core, 1/8 inch cell size and 4.0 lb/ft^3 density. All 350^{O}F cure epoxy-fiberglass, the magnesium and the 0.040, 0.050 and 0.063 aluminum specimens used Phenolic/glass (HRP) core 1/8 inch cell size and 4.0 lb/ft^3 density.

The aluminum alloy used for the front face skins was 2024-T3 non-clad alloy. The epoxy-fiberglass specimens used an "E glass" fabric except for specimen series 24-X which used "S glass". A 120 fabric prepreg (0.0045 inch nominal thickness per ply) was used for the 13.5 mil skins and a 181 fabric prepreg (0.0010 inch nominal thickness per ply) was used for all other epoxy fiberglass skins. The epoxy-graphite skins used 2 ply skin, 0.020 inch nominal, total thickness.

A perforated aluminum back face was used on all specimens except the epoxy-graphite series. The material used was 2219 aluminum alloy, 0.020 inch,

TABLE 2-1

THERMAL EXPOSURE SPECIMENS

TABLE 2-1

THERMAL EXPOSURE SPECIMENS

SPECIMEN DESIGNATIONS	SKIN MATERIAL AND THICKNESS (MILS)	SKIN OR ADHESIVE CURE TEMP. (°F)	COATING SYSTEM MATERIALS/COLOR/THICKNESS (MILS)		
			PRIMER	INTERMEDIATE COATING	TOPCOAT [1]
1-1 thru 1-12	ALUMINUM/12	250	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/5.2
2-1 thru 2-12	ALUMINUM/16	250	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.7
3-1 thru 3-12	ALUMUNUM/20	250	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/5.5
4-1 thru 4-12	ALUMINUM/25	250	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.9
5-1 thru 5-12	ALUMINUM/32	250	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/5.1
6-1 thru 6-12	ALUMINUM/12	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/3.9
7-1 thru 7-12	ALUMINUM/16	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.9
8-1 thru 8-12	ALUMINUM/20	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.2
9-1 thru 9-12	ALUMINUM/25	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.2
10-1 thru 10-12	ALUMINUM/32	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.4
11-1 thru 11-12	EPOXY-FIBERGLAS/13.5	350	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.4
12-1 thru 12-12	EPOXY-FIBERGLAS/20	350	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.4
13-1 thru 13-12	EPOXY-FIBERGLAS/30	350	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.5
14-1 thru 14-12	EPOXY-FIBERGLAS/13.5	250	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.4
15-1 thru 15-12	EPOXY-FIBERGLAS/20	250	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.4
16-1 thru 16-12	EPOXY-FIBERGLAS/30	250	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.4

1 ENAMEL IS MIL-C-83286 ALIPHATIC POLYURETHANE, LAQUER IS MIL-L-81352 ACRYLIC

TABLE 2-1 (CONTINUED)

TABLE 2-1 (CONTINUED)

SPECIMEN DESIGNATIONS	SKIN MATERIAL AND THICKNESS (MILS)	SKIN OR ADHESIVE CURE TEMP. (°F)	COATING SYSTEM MATERIALS/COLOR/THICKNESS (MILS)		
			PRIMER	INTERMEDIATE COATING	TOPCOAT [1]
17-1 thru 17-12	EPOXY-FIBERGLAS/30	250	ASTROCOAT/0.5	ASTROCOAT 8001/WHITE/ 14.3	ASTROCOAT 8003/BLACK/ 1.7
18-1 thru 18-12	EPOXY-FIBERGLAS/30	250	ASTROCOAT/0.5	NONE	ASTROCOAT 8000/BLACK/ 15.0
19-1 thru 19-12	EPOXY FIBERGLAS/30	250	ASTROCOAT/0.5	ASTROCOAT 8001/WHITE/ 16.0	ASTROCOAT 8004/WHITE/ 2.0
20-1 thru 20-12	EPOXY-FIBERGLAS/30	250	NONE	FLUOROCARBON ELASTOMER /WHITE/11.8	FLUOROCARBON ELASTOMER/ BLACK/1.0
21-1 thru 21-12	EPOXY-FIBERGLAS/30	250	NONE	NONE	FLUOROCARBON ELASTOMER/ WHITE/12.4
22-1 thru 22-12	EPOXY-FIBERGLAS/30	250	MIL-P-23377/0.6	NONE	LACQUER/BLACK/4.2
23-1 thru 23-12	EPOXY-FIBERGLAS/30	250	POLYSULFIDE/2.0	NONE	ENAMEL/BLACK/3.2
24-1 thru 24-12	EPOXY-"S" FIBERGLAS/50	350	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.7
25-1 thru 25-12	ALUMINUM/40	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.9
26-1 thru 26-12	ALUMINUM/50	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/4.6
27-1 thru 27-12	ALUMINUM/63	350	MIL-P-23377/1.0	NONE	ENAMEL/BLACK/5.0
28-1 thru 28-12	MAGNESIUM/32	350	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.6
29-1 thru 29-12	EPOXY-GRAPHITE/20	250	MIL-P-23377/0.8	NONE	ENAMEL/BLACK/3.9

1 ENAMEL IS MIL-C-83286 ALIPHATIC POLYURETHANE, LACQUER IS MIL-L-81352 ACRYLIC

nominal thickness, with 0.050 inch diameter holes located on 0.15 inch centers in both directions. The perforated backface material was selected so that the vacuum "chuck", used to hold specimens in the quartz lamp/wind tunnel facility, would apply the vacuum on the front face and thus reduce the probability of the front face being completely delaminated and drawn into the wind tunnel.

The coating materials are identified in Table 2-1 mostly by generic or trade names. More specific identifications are as follows:

MIL-P-23377 is an epoxy primer used as a standard primer on exterior surfaces of Air Force aircraft.

"Enamel" is MIL-C-83286 aliphatic polyurethane enamel. This is the current "standard" exterior enamel for Air Force equipment.

Astrocoat materials are components of rain erosion resistant coating systems meeting either MIL-C-83231 or MIL-C-83445. Astrocoat 8000 is a black erosion coating meeting MIL-C-83231. 8003 is a black anti-static coating also meeting MIL-C-83231. 8001 is a white (yellowing) material meeting MIL-C-83445 and 8004 is a white (non-yellowing) topcoat applied over the 8001 to complete the MIL-C-83445 coating system.

The fluorocarbon systems, either all white or black (anti-static) over white are experimental rain erosion resistant - high temperature resistant coatings developed by the Air Force and manufactured by CAAP Co., Inc.

The lacquer coating conforms to MIL-L-81352; is applied over MIL-P-23377 epoxy primer; and was the previous "standard" Air Force coating material for exterior aircraft surfaces.

With the exception of some of the rain erosion resistant coatings, all the coatings were black. The black color was selected to maximize thermal absorptivity. This was necessary because the thermal exposure facility has a limited thermal flux capability and to keep exposure times within practical limits, a relatively high absorptivity surface was required. Previous data has shown that color, in itself, has little effect on the thermal response of coating materials.

All specimens were layed up and cured in sheets approximately 15 x 20 inches in size, one sheet for each skin or paint system variation. The "front" face of each sheet was then surface treated or cleaned in accordance with standard aircraft practice and painted as detailed in Table 2-1. Each sheet was then cut to provide 12 each $4.500^{+0.000}_{-0.020} \times 3.970^{+0.000}_{-0.020}$ inch specimens.

3.0 SIMULATED NUCLEAR THERMAL EXPOSURES OF SPECIMENS

The test specimens were thermally exposed using the Air Force Materials Laboratory Quartz Lamp Bank-Wind Tunnel Facility. Figure 3-1 is a plan view sketch of the specimen area of this facility.

3.1 TEST METHOD

The Quartz lamps, reflector and cooling air blower (QLB) were mounted on a cart. Quartz lamp voltage was held constant and radiant thermal flux at the specimen was varied by distance of the QLB from the specimen, i.e. peak flux increases as the QLB is moved closer to the specimen. Total fluence was adjusted and controlled by (1) distance of the QLB and (2) by total exposure (power on) time.

Thermal exposures levels were selected to obtain at least four levels of thermal damage for each specimen configuration, i.e.

1. Threshold damage to the paint
2. Severe damage to the paint
3. Threshold damage to the substrate
4. Severe damage to the substrate

Exposures of each specimen configuration were started at a relatively low fluence level and increased as necessary to obtain at least the above 4 levels of damage. Exposures were not, however, increased above approximately 200 cal/cm^2 in that this is considered to be comfortably above a realistic aircraft exposure level, i.e. other effects, such as gust/overpressure damage, tend to become dominant before the thermal reaches the 200 cal/cm^2 level.

Studies of flux intensity over the specimen area show considerable nonuniformity. The center of the specimen face always has the highest flux, with lower, but not necessarily uniformly lower, fluxes toward the specimen edges. Toward the upper flux limit, the center area peak flux is approximately 30% higher than the lowest reading near the edge of the specimen. At lower peak fluxes the difference between the center (high) reading and the lowest edge reading decreases to a minimum of 15%. Appendix I contains several "maps" showing peak flux readings at various points over the specimen face area.

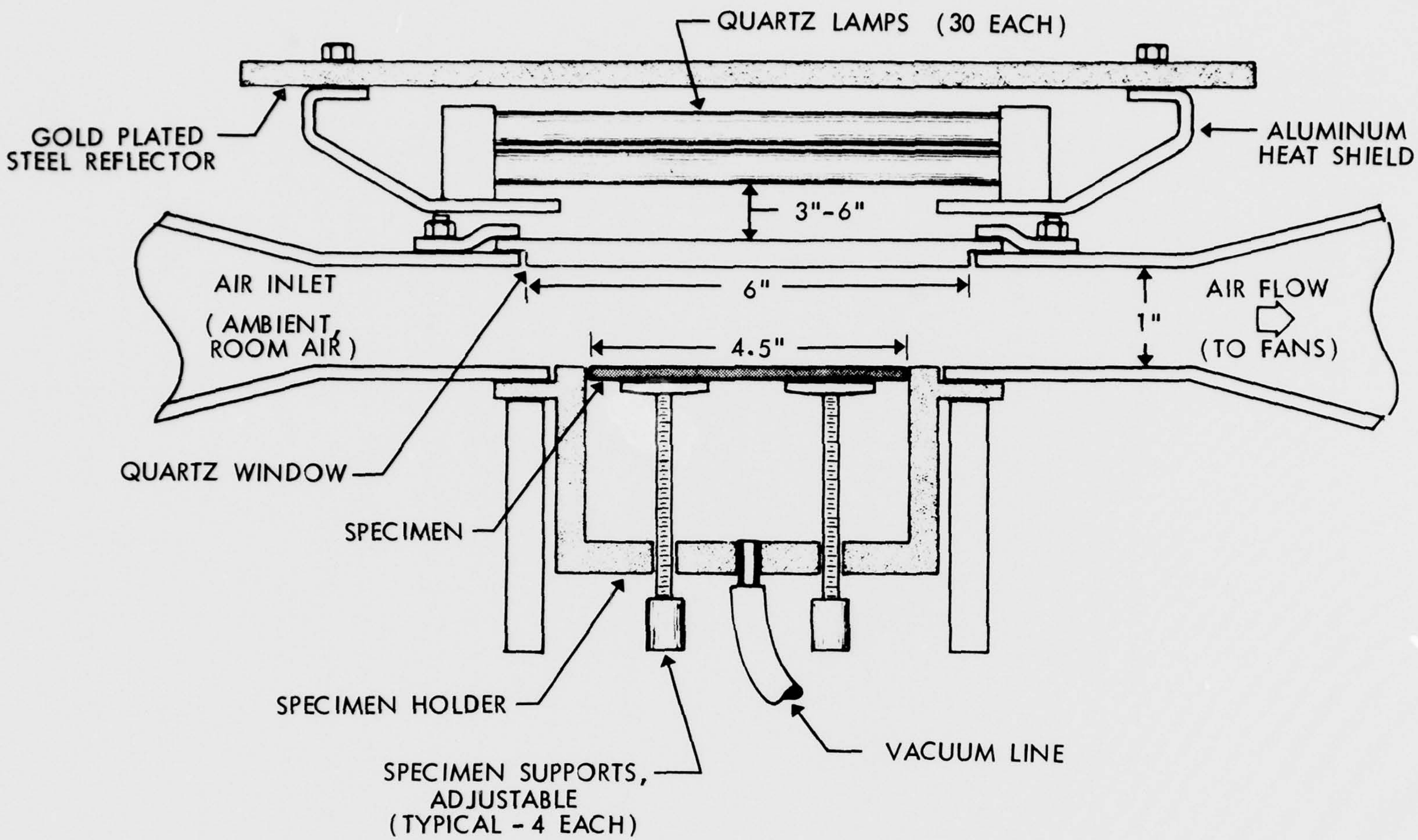


Figure 3-1 Plan View, AFML Quartz Lamp Bank - Wind Tunnel Facility

Except where specifically otherwise indicated, all flux and fluence data in this report are based on center point readings. Fluence data has been obtained by mechanical Planimeter measurements of area under the flux meter vs. time plots.

The wind tunnel is not operated when flux meter readings are being taken.

3.2 THERMAL RESPONSE OF SPECIMENS

A detailed listing of specimen exposures and effects is given in Appendix 2. A summary of the thermal response is given below, and photographs of representative groups of specimens are shown as Figure 3-2 through 3-7.

The "standard" enamel system (MIL-P-23377 epoxy primer plus MIL-C-83286 polyurethane enamel) has a thermal degradation response and thermal capability essentially identical to the lacquer system (MIL-P-23377 plus MIL-L-81352) and the polysulfide primer/enamel (23-X series) coatings. The coating would first lose gloss; it would then form small blisters which would break open to produce a roughened surface and exposue the yellow primer. Next the paint would burn completely off starting in the center of the specimen face and as exposure levels increased, progressing toward the edges of the specimen. This sequence is illustrated in Figures 3-2, 3-3 and part of 3-4.

Among the rain erosion resistant coatings, the all black Astrocoat system was the least thermal resistant. (Figure 3-4, Specimen 18) The black over white system demonstrates the advantage of even a limited reflectivity as ablation progresses. The all white system performs quite comparably to the others when absorbed, rather than incident, heat is considered. The all white system did, however, respond by blistering (probably because of the slower heating rate) rather than the surface liquefaction (only) demonstrated by the all black and the black over white Astrocoat systems. (Figure 3-5, Specimen 19 and 3-6, Specimen 17).

The high temperature resistant fluorcarbon coatings (20-X and 21-X) showed, as would be expected, a considerably higher thermal capability than the Astrocoat. The black over white system showed a surface flow quite comparable to the Astrocoat. However, throughout the ablation the surface stayed dark grey, i.e. significant amounts of the black carbon pigment remained mixed into the ablating surface. (Figure 3-5)

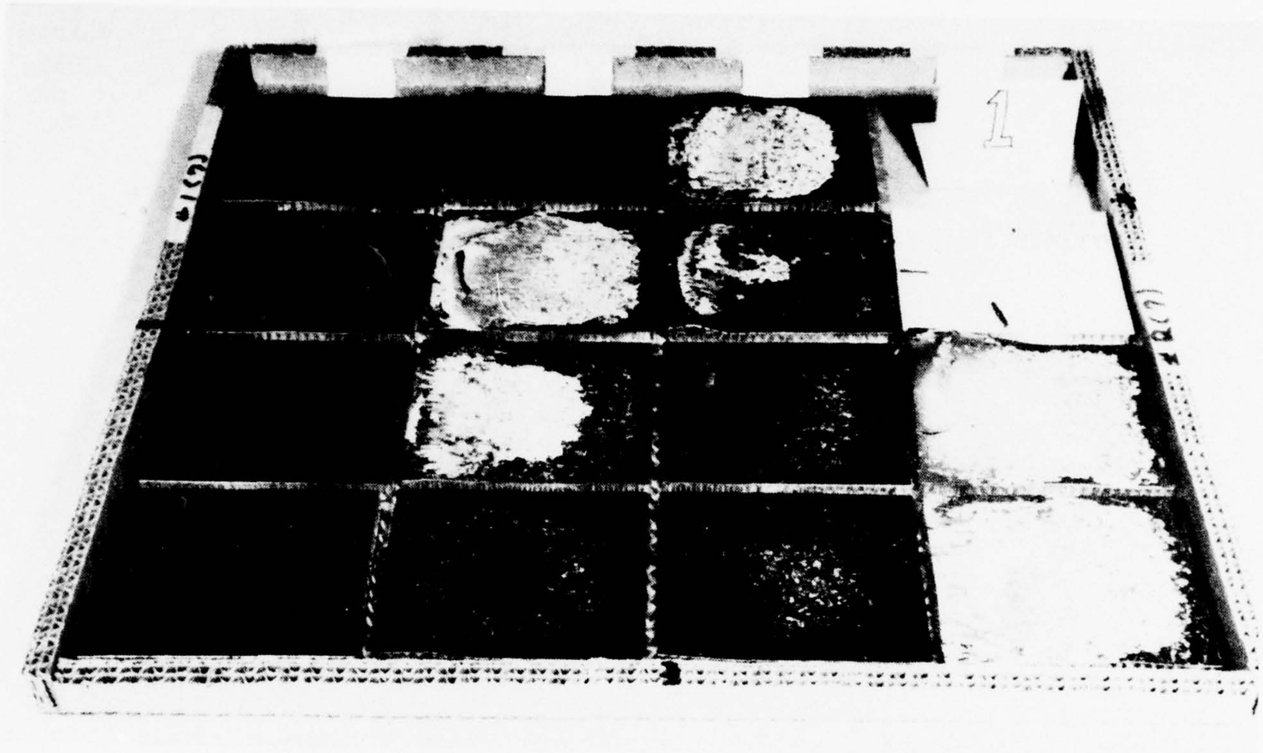


Figure 3-2 Specimens 1 and 2. Black enamel over thin aluminum. The wind flow was from right to left and the exposure increases going from bottom to top, then left to right. (Note: Specimen 1-1 is back to front.)

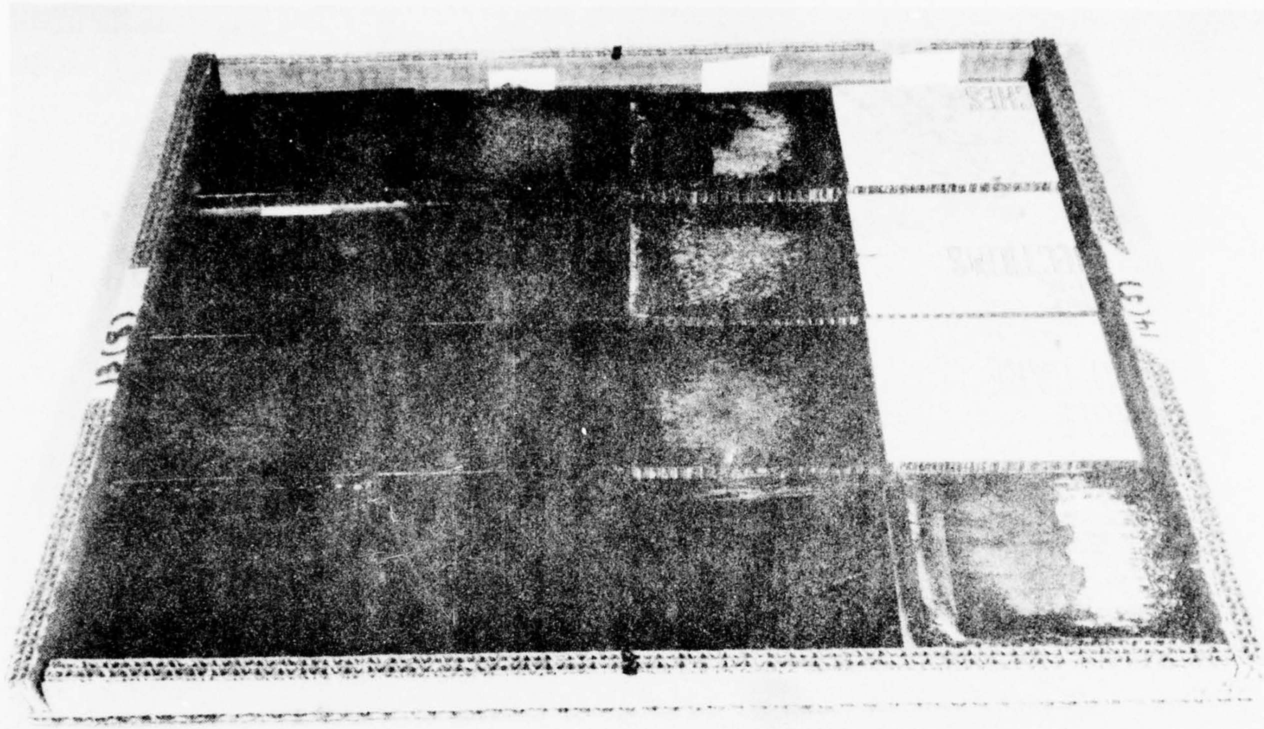


Figure 3-3 Specimens 13 and 14. Black enamel over E-fiberglass. (Sequence is bottom to top, left to right.)

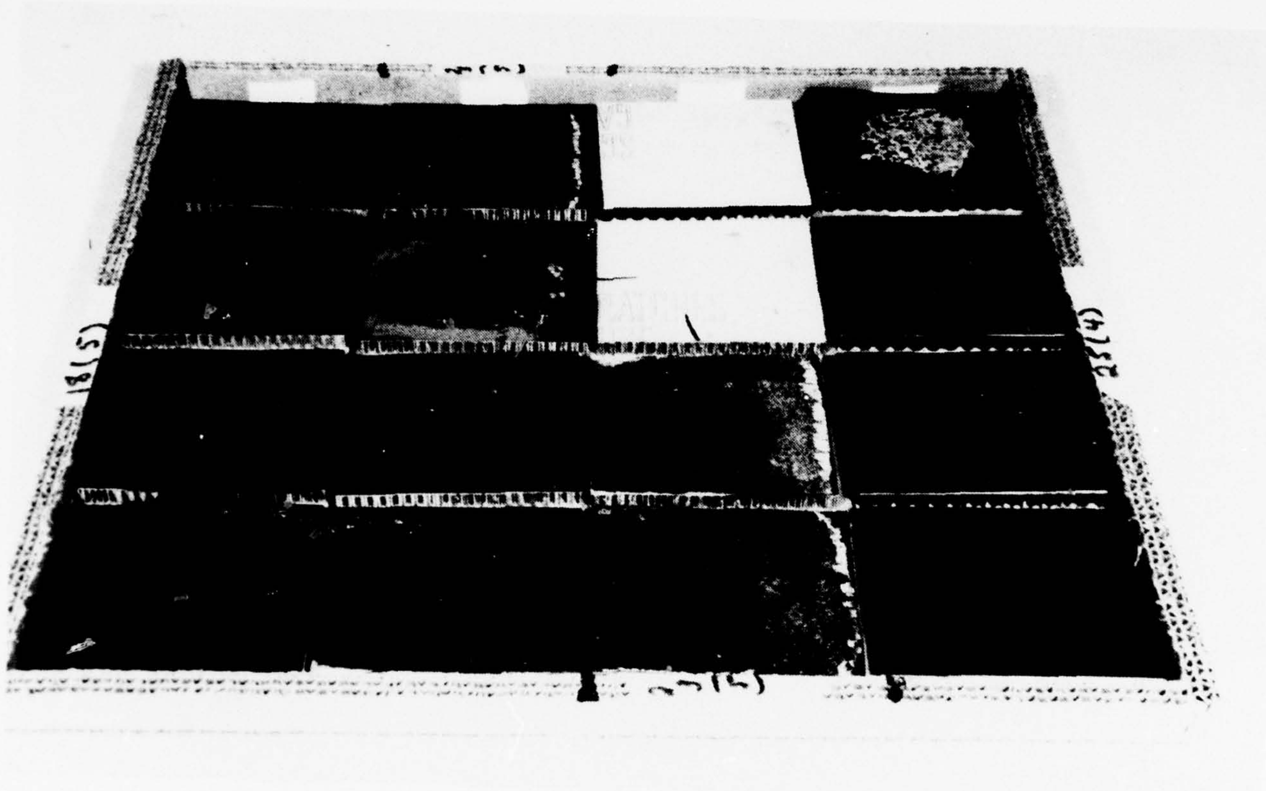


Figure 3-4 Specimens 18, 23 and 25. Numbers 18 and 23 are fiberglass with Astrocoat and enamel respectively, while 25 is a thick aluminum skin (40 mils) with enamel. (Sequence is bottom to top and then left to right.)

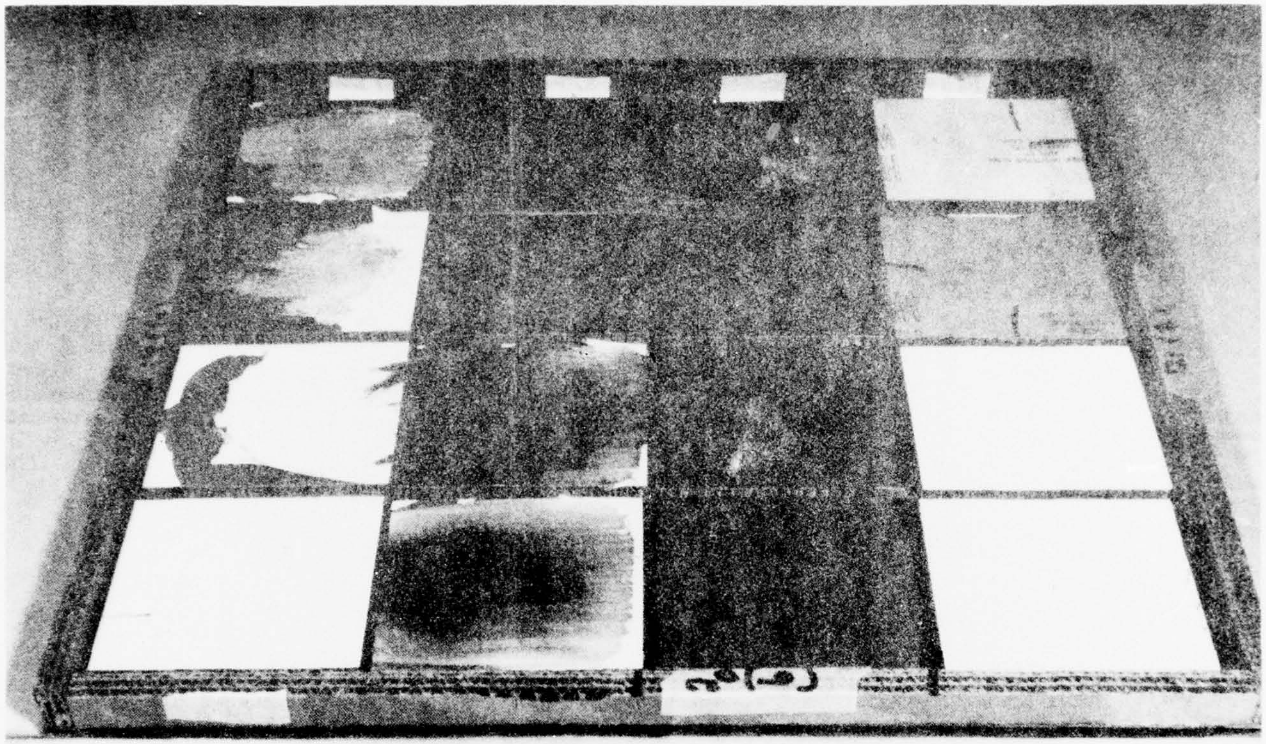


Figure 3-5 Specimen 19, 20 and 21. Fiberglass with Astrocoat and fluorcarbon coatings. (Sequence is bottom to top and then left to right.)

The all white fluorocarbon coating system had a high enough reflectance/heat resistance that the damage threshold was not reached up to the 200 cal/cm^2 fluence level.

The aluminum substrates demonstrated a quite consistent degradation pattern, i.e., the initial effect was warping and delamination of the face sheet which would typically start on the down stream edge. The degree of warpage and the amount (area) of delamination would increase as the exposure levels increased. Eventually an area of skin toward the downstream edge would disintegrate. Typically there was no evidence of liquefaction Rather, the failure edges indicate a crystaline type failure. As aluminum face sheet thickness increased, the damage threshold and total thermal capability increased. (Figure 3-2, and Figure 3-4, specimen 25)

The high temperature (350^{o}F cure) adhesive appeared to increase the delamination threshold, as compared to 250^{o}F cure adhesive. Once past the damage threshold, however, the high temperature adhesive specimens appeared to delaminate faster and the two adhesive (250^{o}F and 350^{o}F cure) systems appeared approximately equal in the total thermal exposure required to produce a severe failure.

The epoxy fiberglass substrates also demonstrated a consistent thermal degradation mode, i.e. the surface would char, char depth would increase, small areas of top ply delamination would appear, the downstream edge of the top ply would start to fray, then up to half of the first ply would disintegrate leaving some fibers still attached at one end. By this point the second ply was typically showing char and some delamination. Total thermal capability increased as total face thickness increased. There was very little difference between the 250^{o}F and the 350^{o}F cure epoxies either in threshold damage exposure level on in total thermal capability. (Figure 3-3, and Figure 3-6)

The "S" glass (24-X series) showed a markedly high threshold for glass disintegration, as compared to "E" glass. The carbon, from the charred epoxy, appears to eventually burn off of the "S" glass leaving the fabric relatively intact and with a fair amount of reflectivity. The "E" glass remains black up to the point where it disintegrates.

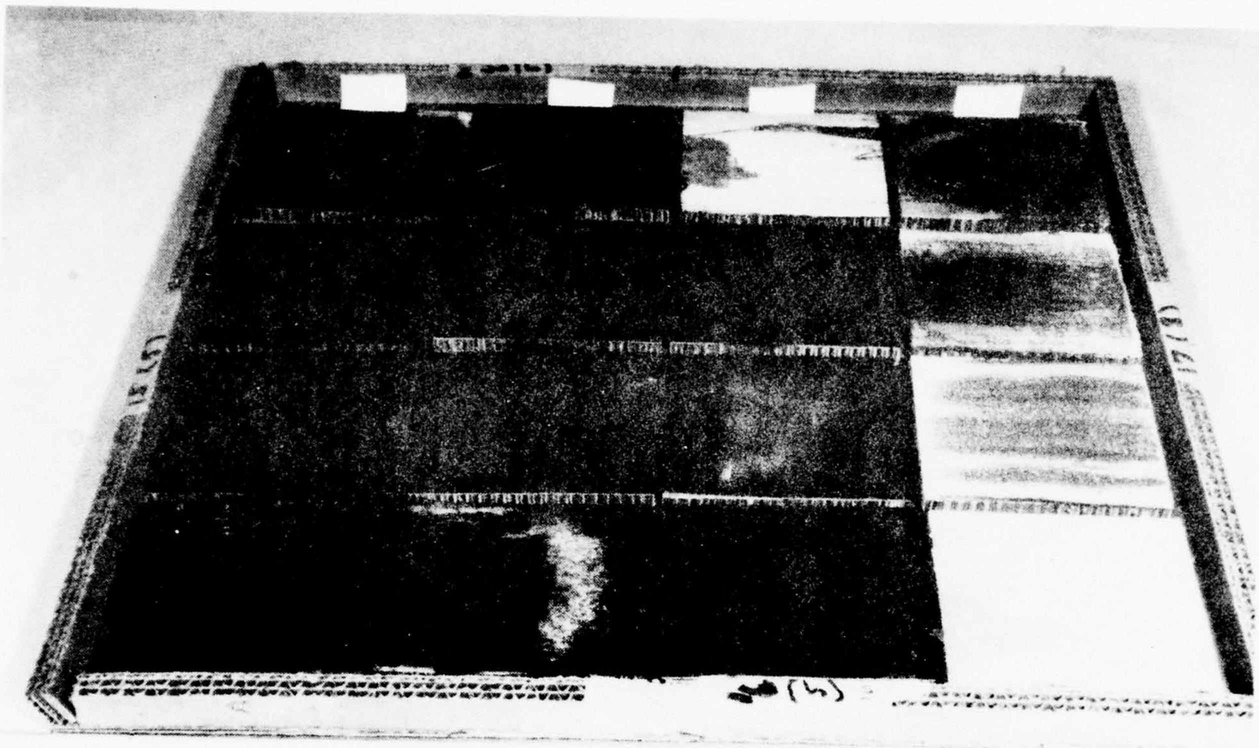


Figure 3-6 Specimens 15 and 16, enamel over fiberglass, and specimen 17, Astrocoat over fiberglass. (Sequence is bottom to top and then left to right.)

The magnesium specimens (series 28-X) appeared to be non-typical of good quality structure, the problem being in the surface treatment. The paint damage threshold was the almost 100% removal of the paint leaving a clean, white metal substrate. The next step in damage progression was loss of the entire face sheet with separation occurring at the metal-adhesive interface. The metal showed no thermal damage, i.e. warping, discoloration, etc.

The epoxy-graphite specimens (Series 29-X) quite rapidly lost epoxy and had major areas of delamination of the first ply. Beyond this point, however, there was little further degradation, i.e. the graphite, even where delaminated, largely resisted disintegration and thus tended to thermally shield the underlying ply. (Figure 3-7)

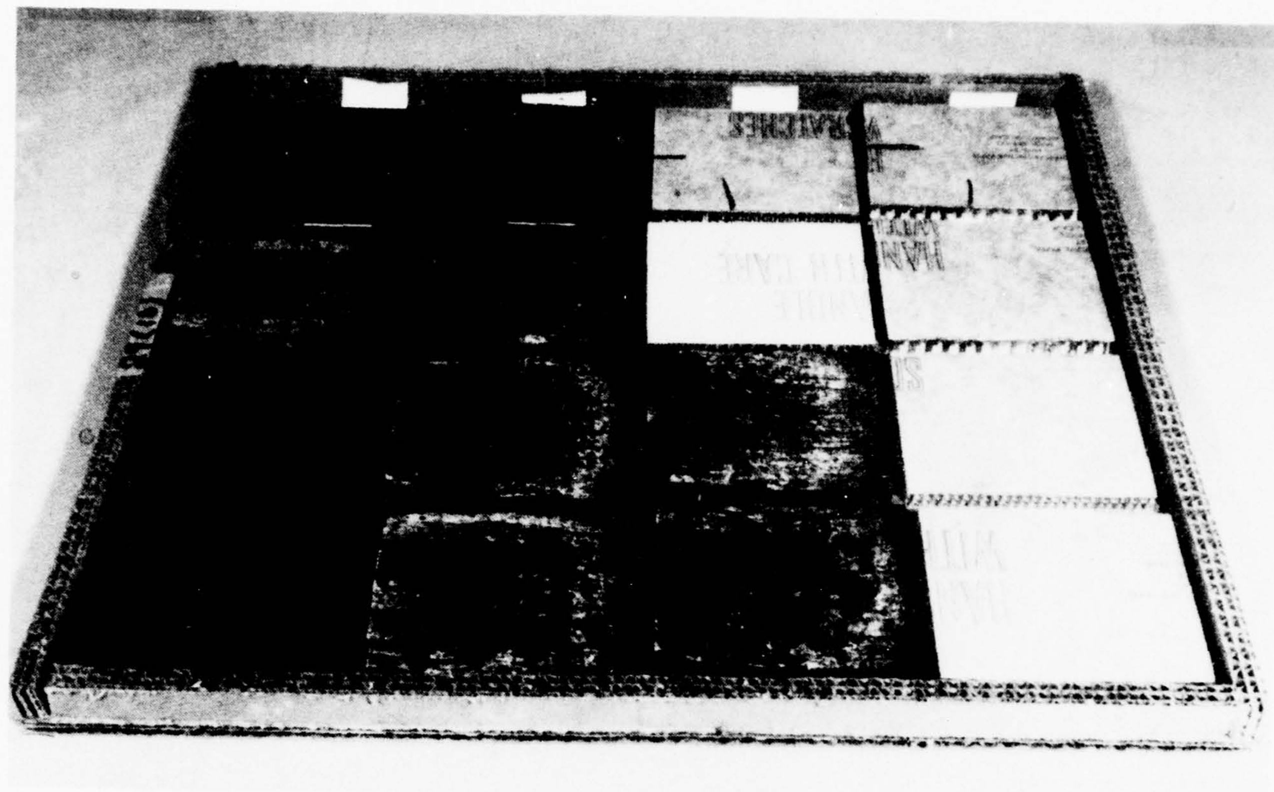


Figure 3-7 Specimen 29. Graphite epoxy with black enamel coating. (Sequence is bottom to top, and then left to right.)

4.0 SKIN FRICTION TEST

4.1 TEST OBJECTIVES

The objectives of this test were to determine an equivalent sand grain roughness of thermally damaged aircraft skin samples, and to present this data in a format usable in standard, existing algorithms for estimating aircraft friction drag. To achieve this, a technique for testing the damaged samples and several reference samples in the Boeing model supersonic wind tunnel was developed. To obtain direct drag readings, a balance and specimen-holding fixture was designed, constructed, and calibrated. Reference samples enabled comparison of test data with previous skin friction measurements to verify the accuracy of the test methods and data reduction.

4.2 TEST EQUIPMENT AND FACILITIES

The tunnel selected for this study was the Boeing model supersonic wind tunnel, a one-tenth scale model of the 4 foot by 4 foot Boeing Supersonic Tunnel. The operating regime of the tunnel is adequate for the test conditions required and easily adaptable to different balance and probe configurations. It is a blowdown type tunnel capable of either subsonic or supersonic flow, with run times ranging from a few seconds to several minutes depending on run parameters. Maximum supply tank pressure is 10 atmospheres. Mach number is controlled by a variable diffuser for subsonic flow, and by changing the contour of the flexible side walls for supersonic flow (Fig. 4-1).

The balance is pictured in Figure 4-2. It consists of an aluminum plate, to which the sample is bolted, which is supported on three flexures. There are two flexures supporting the upstream side of the specimen, neither of which are instrumented. The simple flexure on the downstream side has strain gages attached for the direct drag measurements, and is interchangeable with two other instrumented flexures giving maximum drag levels of 0.25 lb., 1.0 lb, or 5.0 lb. depending on which flexure is installed. The assembly in the center of the balance is an oil filled dashpot designed to reduce vibrations

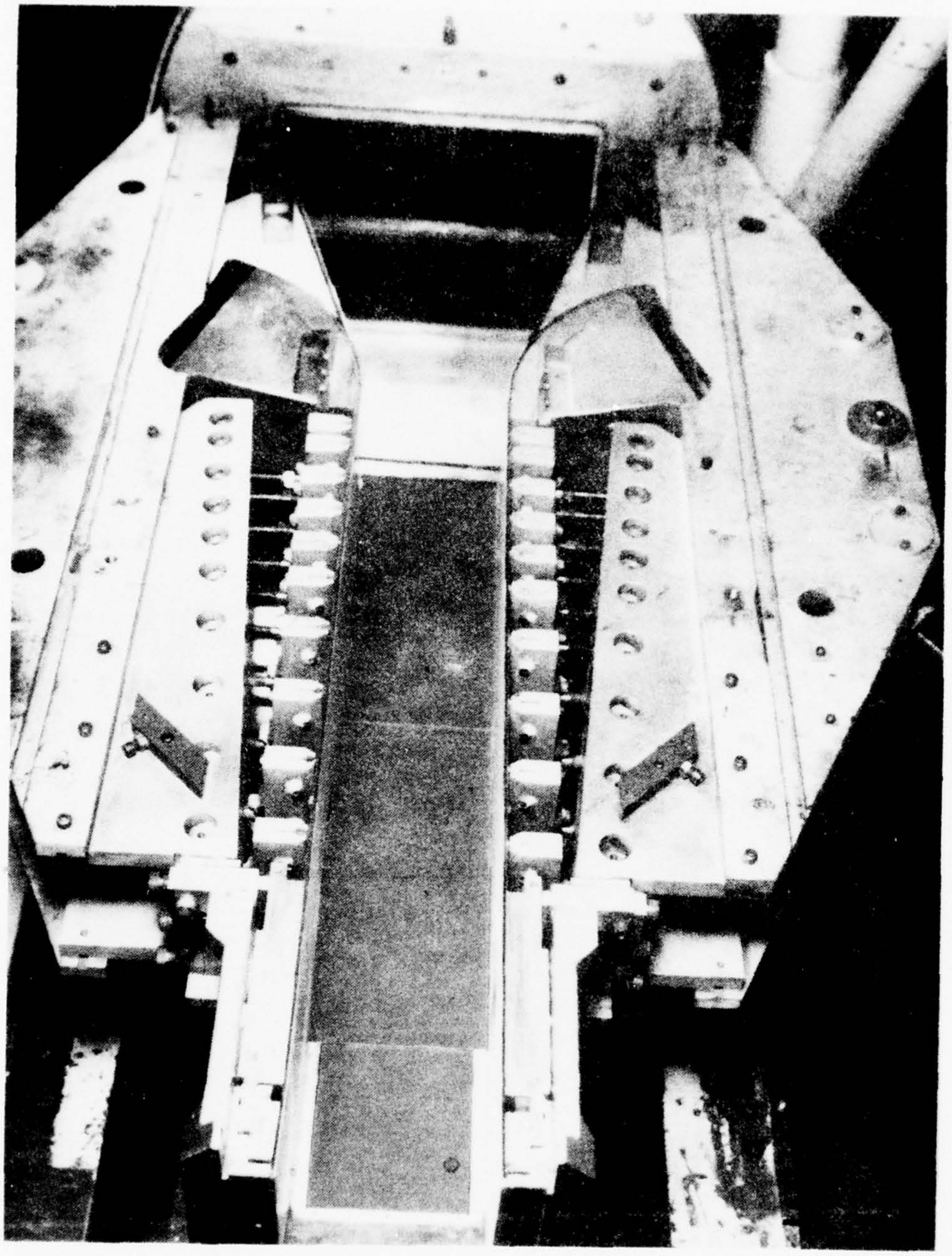


Figure 4-1 Top view of wind tunnel with snadpaper on floor. Test specimen at bottom of figure.

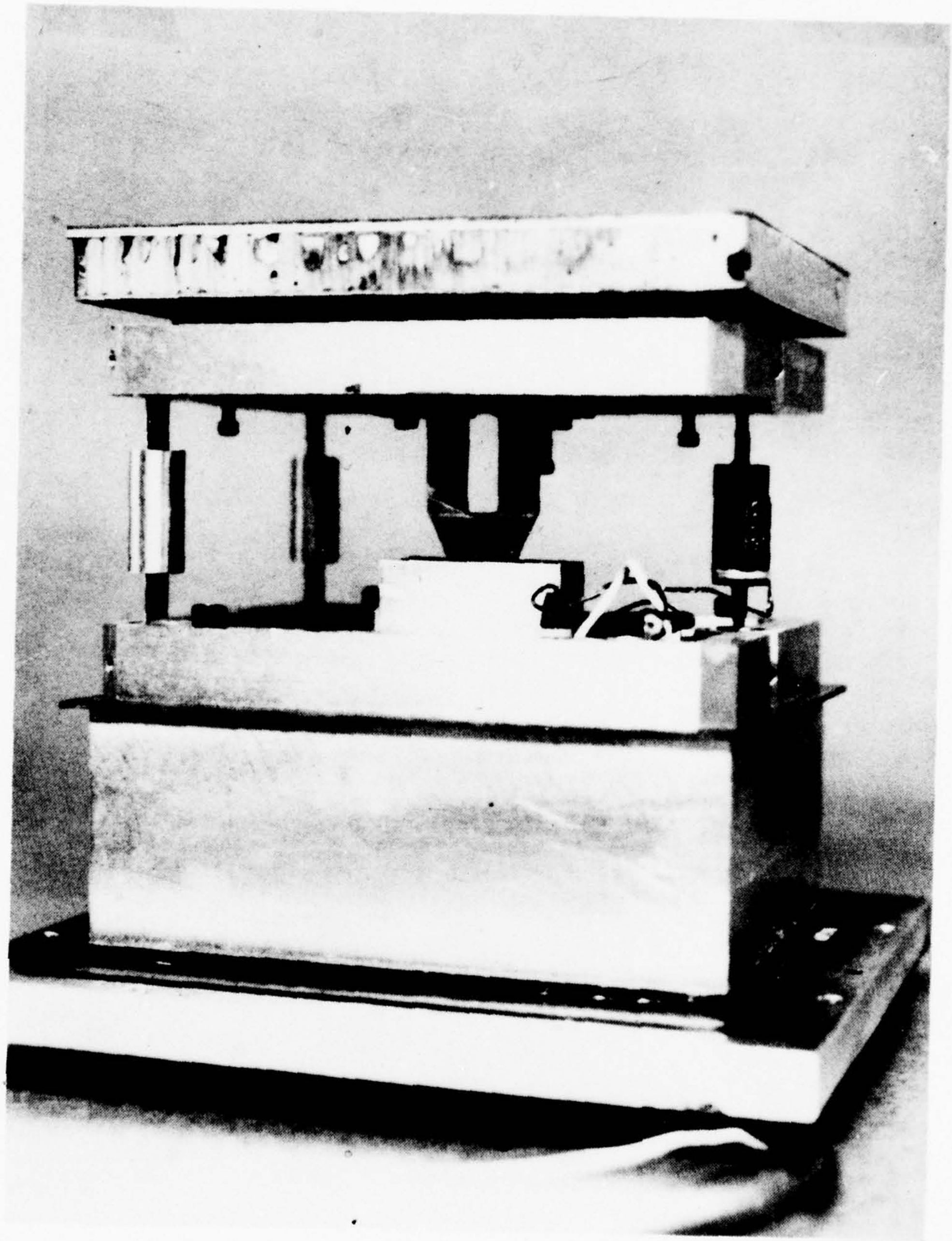


Figure 4-2 Specimen mounted on drag balance. Instrumented flexure is aft.

of the assembly during a run. The rubber O-ring on the base of the balance seals the balance pit and isolates the balance apparatus and tunnel test section from the atmosphere outside the tunnel.

The balance was inserted into a sealed balance cavity in the floor of the test section, and four adjustment screws were then used to make final adjustments of the sample surface relative to the tunnel floor.

To determine the effect of flow through the leading edge and trailing edge gaps between the sample and the balance pit, six pressure taps were installed in the surfaces facing the sample (Figure 4-3), three facing the leading edge, and three facing the trailing edge. Each pair of pressure differentials were then used to find the effect of this flow on the balance readings.

In order to validate test methods through boundary layer velocity profile surveys, a boundary layer probe survey rig was installed in the ceiling of the test section. This included a vertical drive motor and a linear potentiometer for position calibration (Fig. 4-4). Although four probe positions in the test section were available, only one (approximately one inch to the right of centerline near the leading edge of the specimen) was used due to time limitations. The probe dimensions (.027 by .004 inch tip inlet with .001 inch walls) were selected as a compromise between the requirement for a sufficiently small time constant and a probe small enough to minimize self induced disturbances.

The data was recorded on two double channel strip charts and one twelve channel oscillograph. All pressure taps were connected to transducers as close as possible to the tap to reduce stabilization time due to pressure tube volume.

4.3 TEST TECHNIQUE

The wind tunnel test to measure skin friction drag of thermally damaged aircraft skin specimens was conducted in the Boeing model supersonic wind tunnel from March 29 through May 2, 1977. A total of 801 data runs were made during 200 occupancy hours. Measurements were made on 64 different production specimens, plus seven reference models.

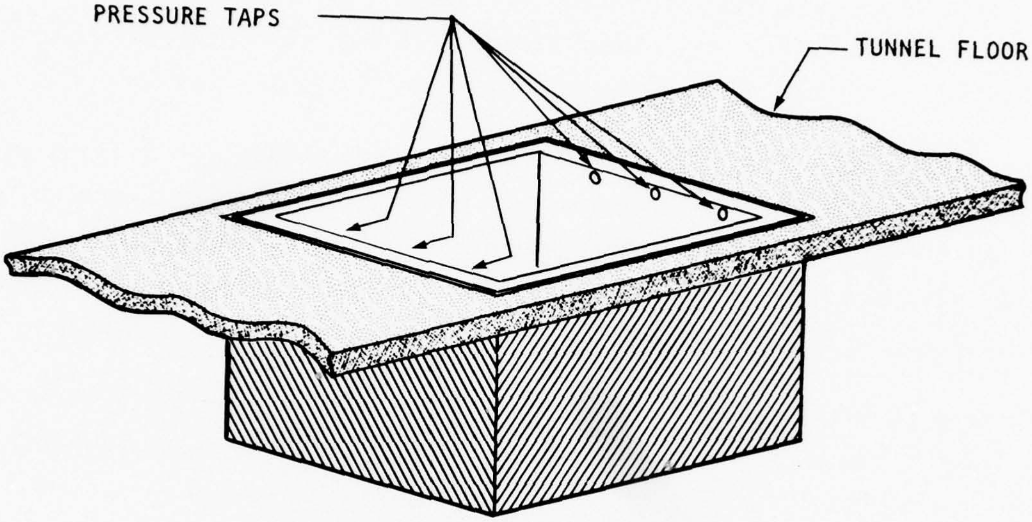


Figure 4-3 Pressure taps in balance box for drag correction factor

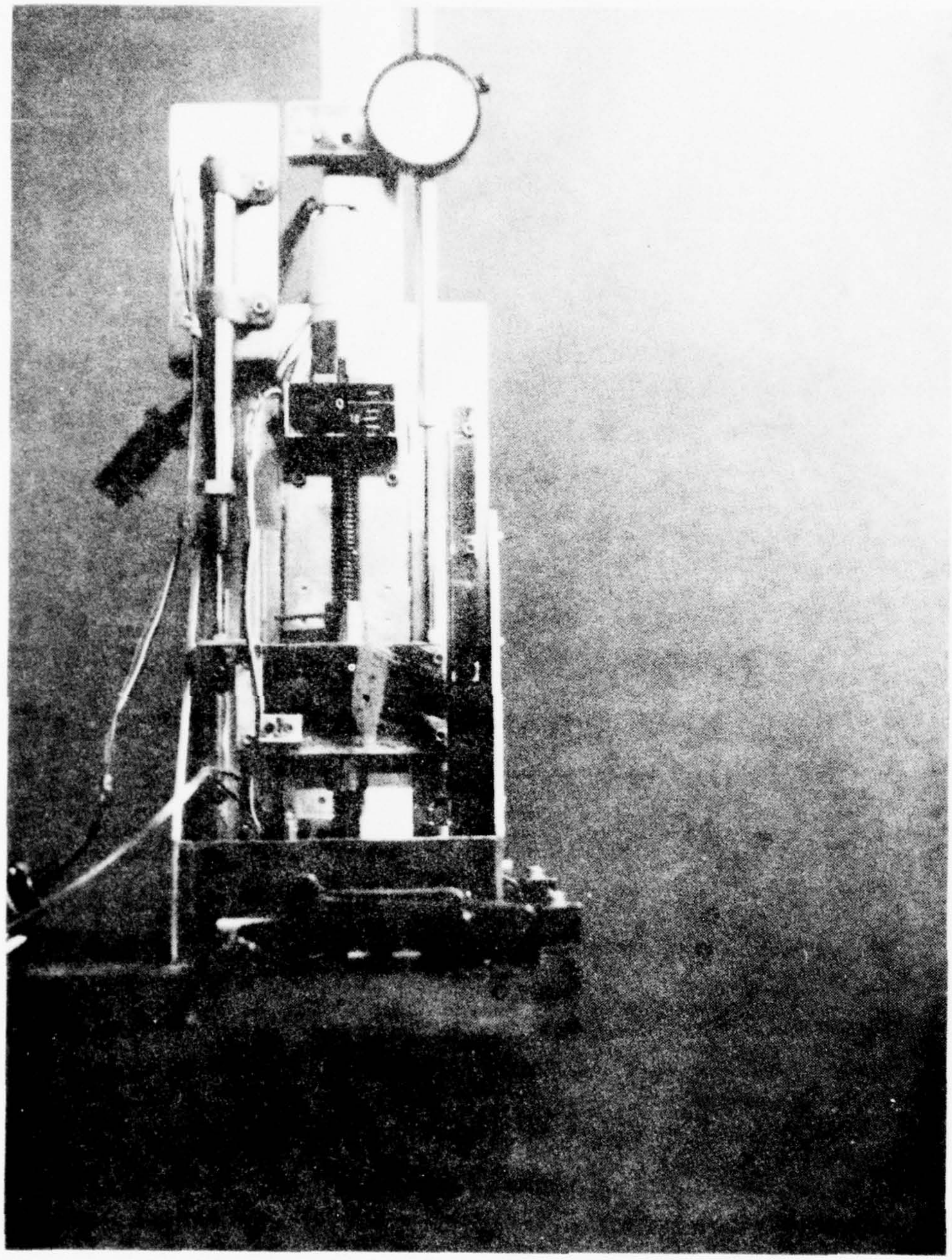


Figure 4-4 Boundary layer probe drive mechanism.

The test specimen surfaces were exposed to the turbulent boundary layer along the tunnel floor. The majority of the data were taken at nominal test section Mach numbers of 0.5 and 0.8 and total pressures of 20, 30, and 35 psia. Additional data were obtained at Mach 1.22 and total pressures of 20, 30, and 40 psia. The corresponding Reynolds numbers per foot ranged from 4 to 13 million.

Prior to testing, the exposed honeycomb on the edges of each specimen was filled and smoothed to reduce turbulence and shear loads in the .02 to .04 inch gaps between the specimen edges and the tunnel floor opening. The 4"x4.5" specimens were attached to a drag balance and inserted into the sealed balance cavity located below the test section floor. Four adjustment screws were then used to accurately adjust the specimen surface to the tunnel floor level. Test data showed that minimum drag was obtained by adjusting upper roughness surface to tunnel floor level, and that there was virtually no allowable tolerance (less than .001") in this setting: drag increased significantly with any vertical displacement of the specimen, either above or below test section floor level.

Initial runs showed large drag variations which were traced to mass flow circulation, into the balance pit at the trailing edge gap and out into the boundary layer at the leading edge gap, removing free stream momentum and increasing skin friction. This problem was solved by sealing the leading and trailing edge gaps. The most effective sealing method was strips of .002 flutter mylar covering the gaps and taped on the upstream sides (Fig. 4-5). Balance interference was insignificant and the durability of the strips was sufficient for the test conditions. The mylar strips were cut so that they were long enough to cover the gaps and not be drawn into them, but not so long that they would flap in the airstream and increase drag levels.

The trailing edge tape showed a consistent drag reduction of approximately 40% to 50%, and the leading edge tape showed no change. This indicates the direction of circulation around the specimen, as the tape could not be as effective stopping flow out of the cavity as it would be stopping flow in. Leading edge tape (attached to the tunnel floor) was used throughout the test program to compensate for the surface area of the trailing edge tapes (attached to the test specimens).

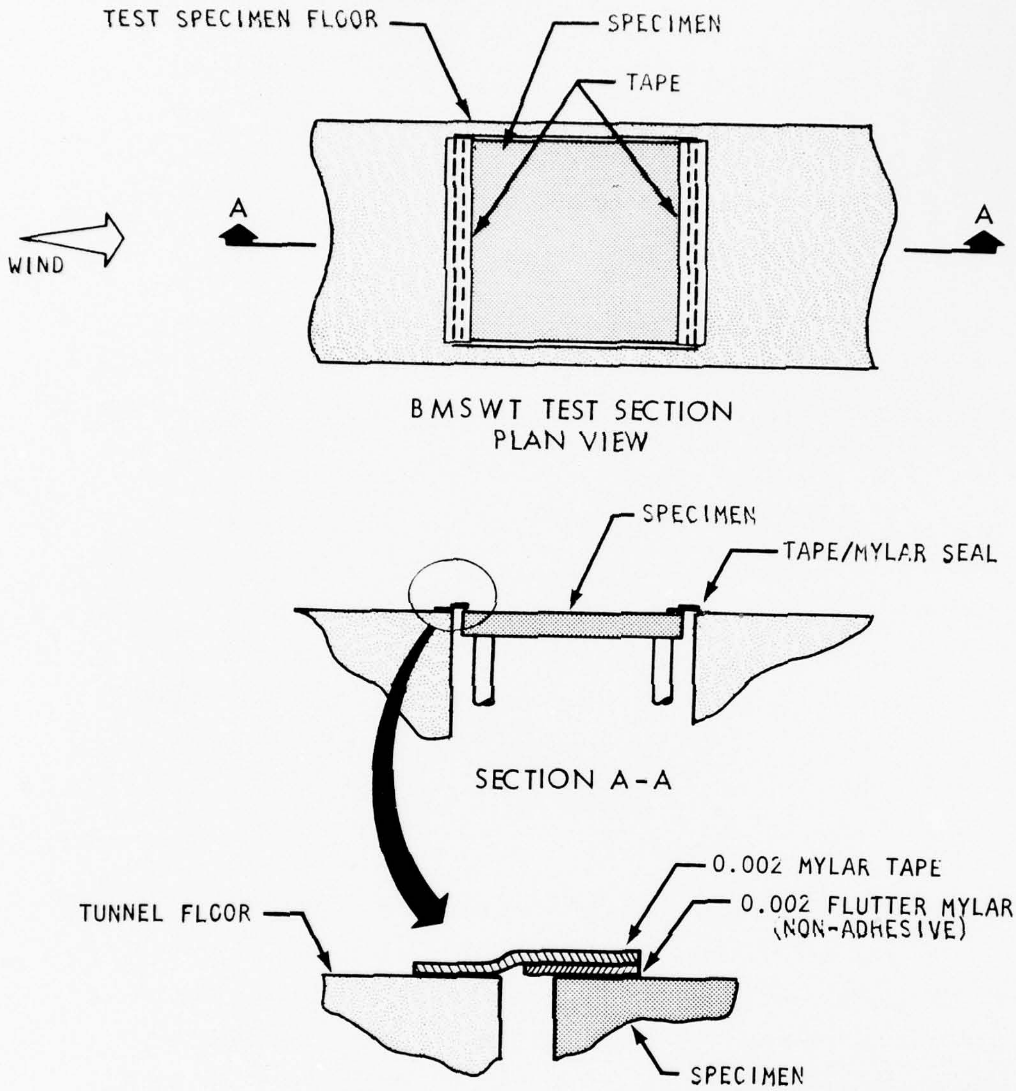


Figure 4-5 Leading/trailing edge gap seals

The pressure ports (Fig 4-3) can then be used to correct the balance data for any additional circulation due to tape "leaks" or turbulence due to the uncovered gaps on the sides of the specimen. These three Δp's were averaged, multiplied by the end area of the sample and subtracted from the balance drag measurement to obtain corrected skin friction drag. Repeat runs with reference and production samples showed this method to be very reliable in providing consistent corrected data.

The unsealed side gaps probably had some effect on the drag measurements. However, it was impossible to seal these gaps, and since the test results with the reference specimens generally agreed with classical skin friction standards, no additional corrections for side gaps were attempted.

Once the specimen height was adjusted and the gaps were taped, a test run sequence was made. A standard sequence for subsonic runs was M = .5 for total pressures of 20, 30, and 35 psia followed by M = .8 at the same total pressures. All drag runs were made with the boundary layer probe raised to the ceiling as the probe was found to affect balance data when near the sample. Tapes were checked frequently and often had to be changed during the run sequence, but several repeatability checks proved that a tape change did not significantly affect data.

Boundary layer velocity profile sweeps were made separately from the drag data runs. In most cases, the probe was lowered to within .001 inches from the floor, the tunnel was started, and the probe was raised. The probe was t-o small to allow pressure stabilization during a constant sweep, so the boundary layer had to be swept in increments, allowing the pressure to stabilize before moving the probe. At high total pressures the run time was too short to allow a full boundary layer sweep, so the tunnel was shut down until the supply tank pressure built up, then restarted. This process was repeated as often as necessary to get through the boundary layer thickness, which ranged from 0.3 inches to 0.6 inches.

Unfortunately, tunnel blockage resulted when sweeps at M - 1.22 were attempted, so none were taken.

4.4 DATA REDUCTION

Data reduction began as s-on as each run was terminated. For all runs (balance data or boundary layer sweep), the experimental values for total and static pressures were reduced to actual test section Mach number. From this value, dynamic pressure and Reynolds number per foot were determined. Reynolds number is a function of total temperature, but the variation of Reynolds number is a function of total temperature, but the variation of Reynolds number over the temperature range encountered during the test was found to be negligible, so total temperature was assumed a constant.

For drag data runs, three pressure differentials were read from the pressure taps described in section 4.2. There pressures were averaged and multiplied by the end area of the sample being tested, and this drag correction was subtracted from the direct balance reading to give a corrected drag value. Drag was then changed into coefficient form by the equation:

$$C_{f_c} = \frac{D}{qS}$$

where q is dynamic pressure and S is the surface area of the sample (18 sq. in.)

Incompressible values were found from this compressible drag coefficient through two formulas. For the smooth plate sample, the compressibility formula was:[1]

$$\frac{C_{f_i}}{C_{f_c}} = (1 + .115\, M^2)^{.75}$$

and for the rough plate, it was:[2]

$$\frac{C_{f_i}}{C_{f_c}} = 1 + r\, \frac{\gamma - 1}{2}\, M^2$$

where γ = 1.4 and r, the recovery factor is assumed to be 0.86.

---

[1]N.B. Cohen "A method for computing turbulent heat transfer in the presence of a streamwise pressure gradient for bodies in high speed flow" NASA Memo 1-2-59L, 1959.

[2]H.W. Liepmann and F.E. Goddard. "Note on the Mach number effect on the skin friction of rough surfaces". J. Aero Sci. 24 784 (1957).

These incompressible, corrected skin friction drag coefficients were used in the final analysis.

Getting momentum thickness from boundary layer surveys, plotting total pressure against height above sample, was a more complex procedure. First, free stream values had to be found for Mach number, temperature, density and speed of sound to get velocity (U_e).

For each point through the boundary layer new values for boundary layer Mach number, velocity, temperature, and density were found to get each increment of momentum layer thickness:

$$\eta = \frac{\rho}{\rho_e} \frac{U}{U_e} \left(1 - \frac{U}{U_e}\right)$$

A numerical integration formula was then used to integrate these values to get momentum thickness:

$$\theta = \int_0^{y_e} \eta \, dy$$

where y_e is at the edge of the boundary layer.

The boundary layer profiles taken were divided into .02 inch increments (arbitrarily), giving approximately 20 data points for each boundary layer survey.

Of course since no boundary layer surveys were possible at the chosen supersonic Mach number, no momentum thickness was available for supersonic data analysis.

4.5 ANALYSIS METHOD

The analysis of the test results, and reduction of the thermally roughened skin sample drag measurements to an equivalent roughness height, are based on formulae developed by Schlichting[1].

$$c_f = \left(1.89 + 1.62 \log \frac{X}{k_s}\right)^{-2.5} = \frac{2\theta}{X} \qquad (1)$$

$$c_f' = \left(2.87 + 1.58 \log \frac{X}{k_s}\right)^{-2.5} \qquad (2)$$

where:

c_f = total skin friction coefficient

c_f' = local skin friction coefficient

X = distance from start of turbulent boundary layer

k_s = height of roughness elements (sand)

θ = momentum thickness

These formulae were determined empirically for the completely rough regime, and are based on measurements by Nikuradse on sand roughened pipes. The transposition of pipe flow results to flat plate flow is quite straightforward, but there is a question as to the distribution of roughness in the sand used in Nikuradse's experiments. This question will be considered further in the section below on sand paper test specimens.

However, the major difficulty in directly using Schlichting's expressions is that they are based on surfaces roughened over the complete extent of the turbulent boundary layer flow. The present experimental set-up, as described above, used a rough test sample inserted in the tunnel wall after a significant length of turbulent boundary layer flow over a smooth surface. Thus a method was developed to estimate an "equivalent length" of upstream flow for this rough samples. Figure 4-6 indicates the growth of momentum thickness along the tunnel floor. (grossly exaggerated vertically)

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[1] H. Schlichting, "Boundary Layer Theory", P. 553 4th Ed. McGraw Hill, 1962

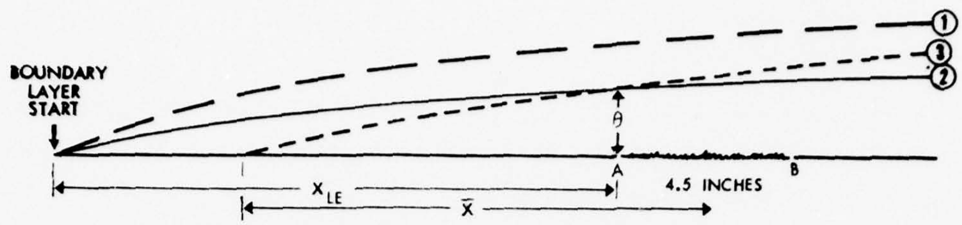

Figure 4-6 Growth of momentum thickness

The test plate (length 4.5", roughness k) is distance X_{LE} from the start of the boundary layer. If the tunnel floor upstream were also of roughness k, the boundary layer (and momentum thickness) would grow as indicated by curve 1. However with the smooth floor, it would grow more slowly (curve 2) to the leading edge of the rough plate and then it would grow at a rate governed by its thickness there and the roughness of the plate (curve 3). We assumed that this growth was governed by equation 1 above with an appropriate value of x. This distance was taken to be the distance that would have been needed for flow over a surface of roughness k to have the momentum thickness grow to the actual value measured at the leading edge of the plate.

This procedure is shown in Figures 4-7 and 4-8. Figure 4-7 is obtained from equation 1 and the smooth plate momentum thickness equation (see below, equation 3). Figure 4-8 is plotted from equation (2) with the x obtained from Figure 4-7 for each k_s to match momentum thickness at the test plate leading edge. Figure 4-7 is entered with the measured value of θ for the smooth plate. This value of θ is read across to the curve for the particular value of K_s to be plotted. The x value is read, and 2.25" is added to get to the mid point of the test plate. This total value is then $\bar{x}$, the (average) equivalent distance used to obtain a point in Figure 4-8 from equation 2.

If this analysis method is valid Figure 4-8 may then be used to give the value of k_s corresponding to the drag on a rough plate for given flow conditions.

The method used to validate this procedure is described below, but, essentially with measured skin friction, sandpaper of known roughness was used, to obtain data points for correlation with curves such as Figure 4-8.

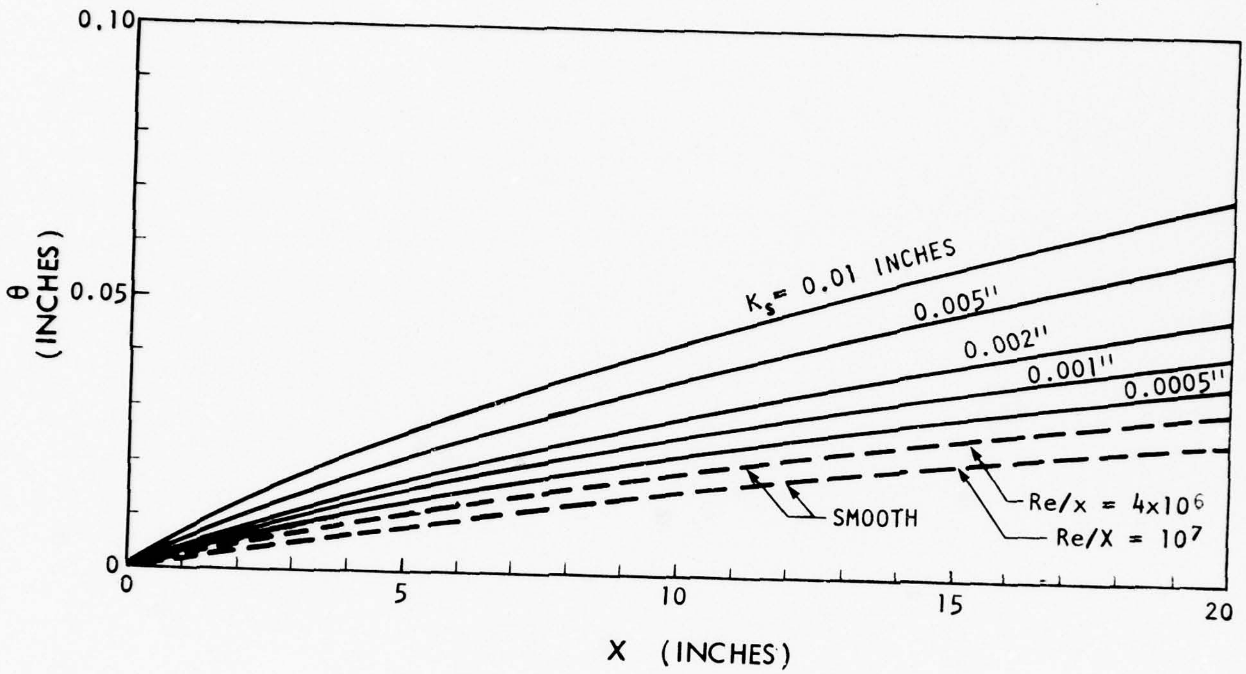

Figure 4-7 Effect of upstream roughness on momentum thickness

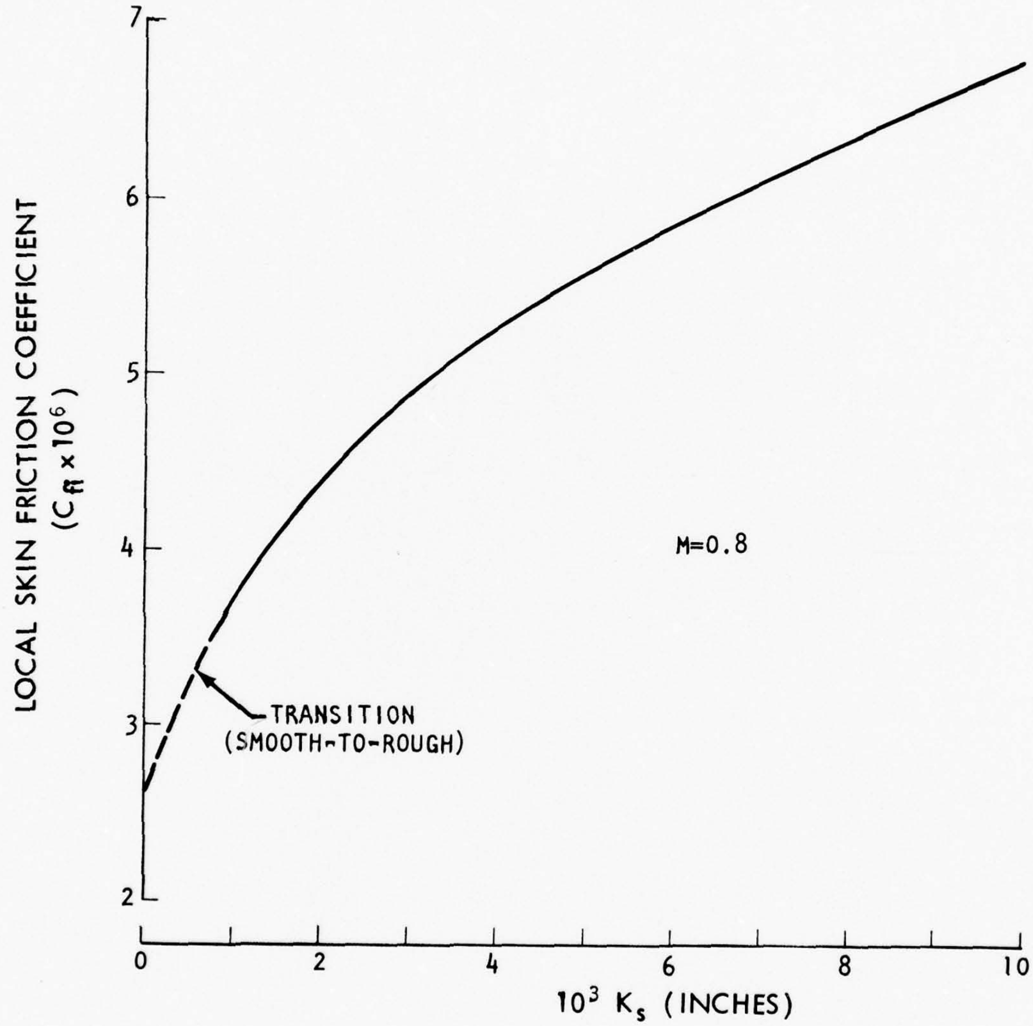

Figure 4-8 Roughness height versus skin friction, based on Schlichting formula and Boeing wind tunnel

4.6 ANALYSIS VERIFICATION

Sandpaper samples of various known grit sizes were used as the basic verification tool for the analytic method described above. In addition sandpaper was glued to the tunnel floor for various lengths upstream of the test specimen to verify the equivalent length concept. Before these tests however an extensive series of tests were run with a smooth test specimen to verify basic test measurement techniques and equipment.

4.6.1 Turbulent Boundary Layer Initiation

The Reynolds number at the test plate varied from about 6×10^6 to 15×10^6 for the subsonic runs. Although this appeared to be high enough to ensure a turbulent boundary layer, trip strips were used to try to fix the transition point. Two trip strips were used, consisting of #100 grit glued to the tunnel floor. The first one was 0.1" wide, 20" upstream from the leading edge and the second one (used for all the production runs) was 0.25" wide, 23" upstream. These strips were located just in the bell-mouth portion of the tunnel.

Boundary layer surveys were made for a series of runs at M = .5 and .8, with tunnel pressures of 20, 30 and 35 psia, and the probe about 0.25" upstream of the test plate. The momentum thicknesses were calculated and are shown in Figure 4-9. The reason for the data scatter is not clear - for example there was no consistent pattern between the trip strip used and momentum thickness. The boundary layer profiles also showed noticeable variation, and it was clear that they were not uniform "1/7 power" profiles, but no attempt was made to fit a logarithmic profile to them. Even after a great deal of care to set repeatable tunnel conditions, and to make sure that the probe was giving good measurements, the variability remained. A possible cause was the valve controlling the flow into the plenum chamber. This valve typically introduces turbulence into the flow with some variable effect on the transition point and the profile of the turbulent boundary layer.

These values of momentum thickness were next used to derive a nominal starting point for the turbulent boundary layer, since it was clear that it was not starting exactly at the trip strip. To do this we used the Schlichting relation (based on a logarithmic velocity profile).

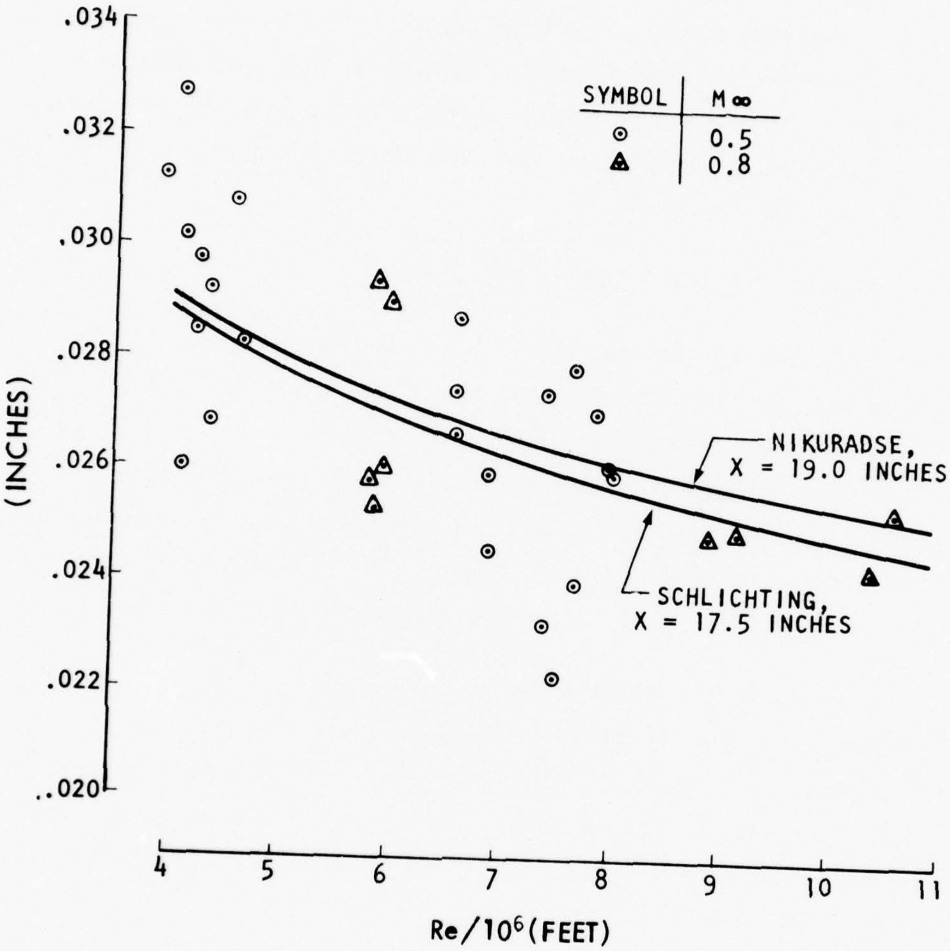

Figure 4-9 Smooth plate boundary layer momentum thickness at leading edge of test specimen

$$\theta = .228X\ (\log Re_x)^{-2.58} \quad [1] \tag{3}$$

Equation (3) was solved for x using measured θ's as an input. The results are shown in Figure 4-10. A nominal starting length of 17.5" was chosen. The momentum thickness using this length is shown in Figure 4-9 and was used as one of the basic curves for obtaining an equivalent length for the rough test specimens. Other formulas for θ could be used, and Nikuradse's experimentally based one was investigated. This gave a length of about 19". However, since Schlichting used equation (3) as the smooth extreme for his rough plate work it was felt appropriate to use his formula.

4.6.2 Skin Friction for the Smooth Plate

The next part of the experimental program was to determine the local coefficient of skin friction for the smooth flat plate. One of the original specimens with the thickest aluminum face sheet (.063") was chosen for this test article and the surface polished before the test. The surface profile was measured (Figure 4-11) at 40 micro-inches peaks after all the tests were completed. The enamelled surface on the painted specimens was a little smoother, with peaks of about 10 micro-inches, but 40 micro-inches was definitely in the smooth regime (about 200 micro-inches) for these tests.

Significant difficulty was initially encountered obtaining results comparable to Schlichting's, but good data were eventually obtained by the use of tape over the leading and trailing edges, and using extreme care in adjusting the height of the specimen. It was found that the height of the sample with respect to the tunnel floor was critical, and this adjustment had to be made to less than .001". A misalignment of the leading edge of about .002" gave results 50% greater than the norm. The results finally obtained are shown in Figure 4-12. The curve drawn on the figure is based on Schlichting's empirical equation for local skin friction coefficient

$$c_f' = (2 \log Re_x - 0.65)^{-2.3} \quad [1] \tag{4}$$

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[1] H. Schlichting. "Boundary Layer Theory", P. 540.

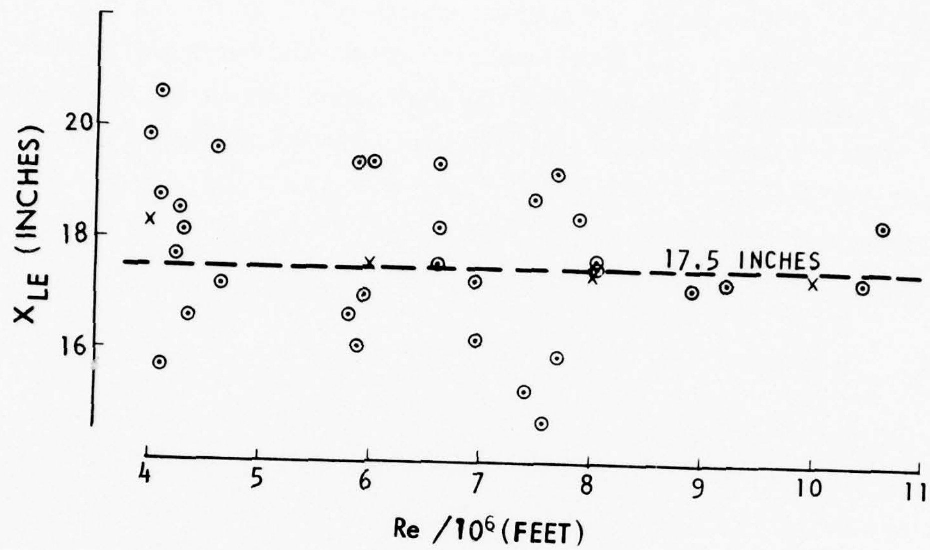


Figure 4-10 Length of turbulent boundary layer, based on measured momentum thickness

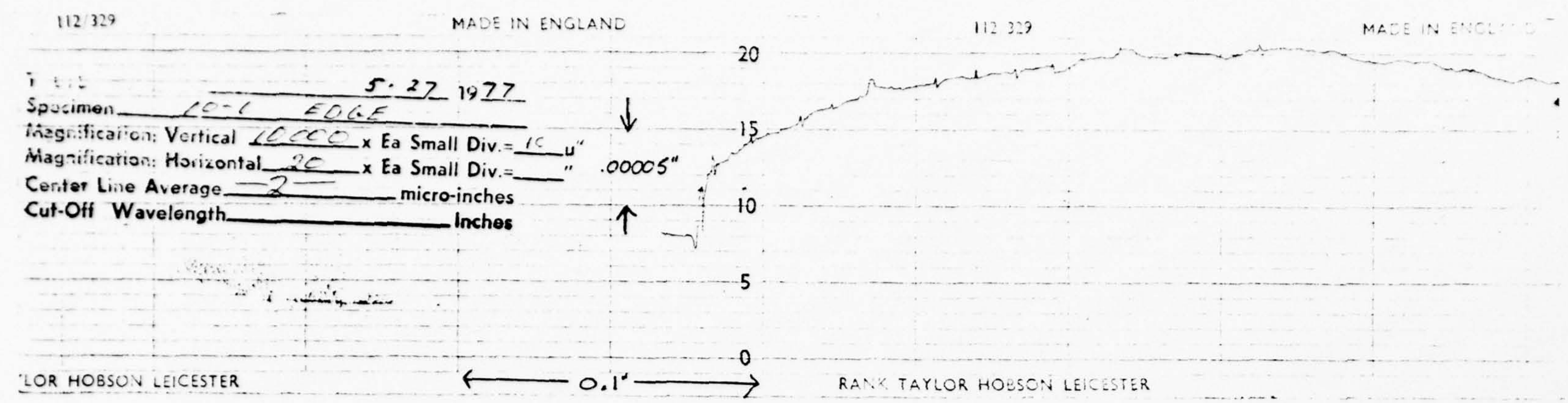


FIGURE 4-11 (a) PROFILE OF SMOOTH ENAMELLED SURFACE (SPECIMEN 10-1)

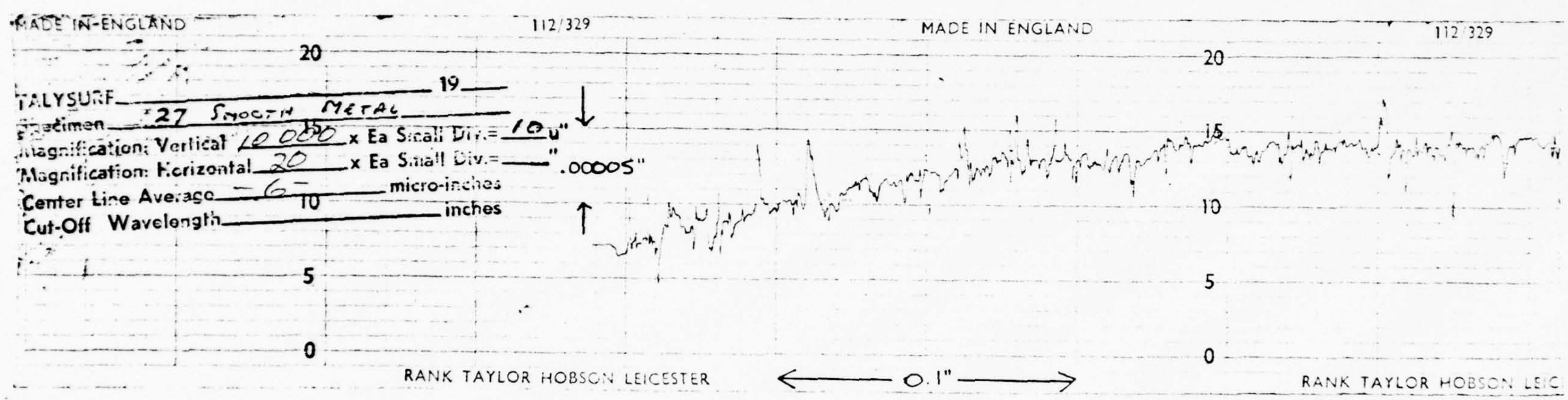


FIGURE 4-11 (b) PROFILE OF SMOOTH ALUMINUM SKIN (SPECIMEN 27)

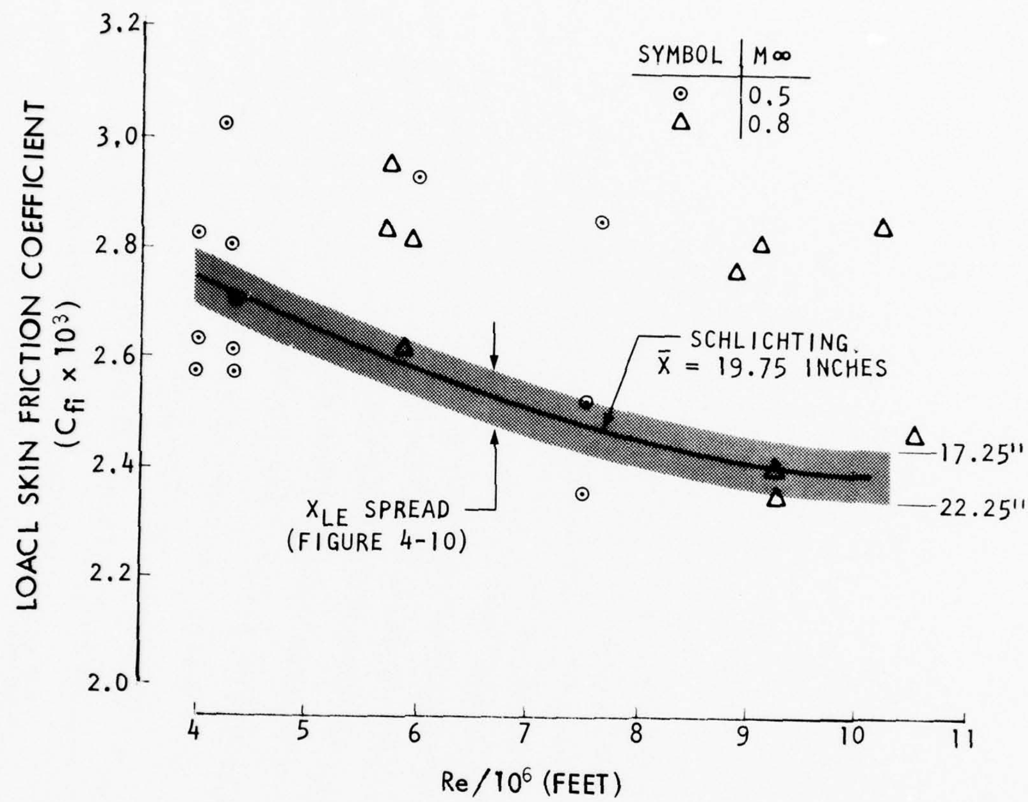


Figure 4-12 Smooth plate, local skin friction coefficient

The upper and lower curves give values for $\bar{x}$ = 17.25" and 22.25", corresponding to the X_{LE} scatter in Figure 4-10 from 15" to 20". It can be seen that this scatter in turbulent boundary layer length is not very significant in predicting c_f'.

The measured data has a scatter of about $\pm$ 7% and appears to have a mean higher than the empirical formula by about 5%. This is to be expected, since both high and low specimen misalignments gave an increase in drag. These results show that the experimental arrangement gave satisfactory agreement with the basic correlation formula for the smooth plate.

4.6.3 Rough Surface Reference Specimens

If the thermally damaged skin specimens had extended from the start of the turbulent boundary layer, no further validation tests would have been necessary. However the specimens were inserted in the tunnel floor with about 17.5" of flow over a smooth wall before the boundary layer met the roughness specimens. A procedure was developed, described in Section 4.5, to account for this by assuming an equivalent length of rough upstream wall to give an identical momentum thickness to that actually occurring. To validate this method, reference specimens of known roughness were used. Using the suggested theoretical method, Figures 4-13 and 4-14 show the relation between local skin friction coefficient and test specimen roughness for a particular Reynolds number and Mach number. Sandpaper of various grit size (100, 220 and 600) was glued to test samples, and the local skin friction coefficients were measured. Results are plotted on Figures 4-13 and 4-14 and show excellent correlation. These figures were then used for the conversion of measured skin friction coefficient to the equivalent sand roughness for use in drag calculations.

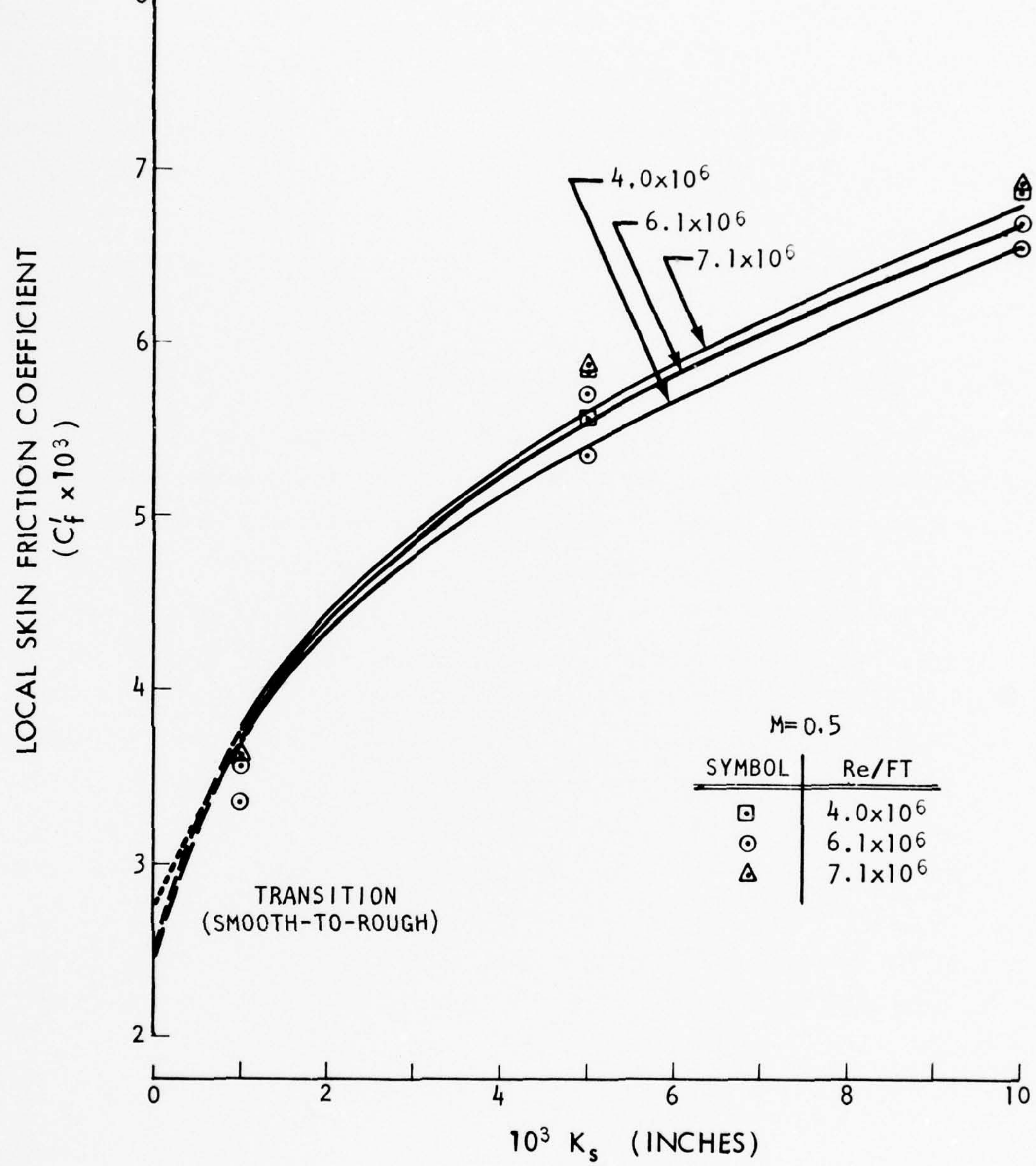


Figure 4-13 Effect of roughness on incompressible skin friction

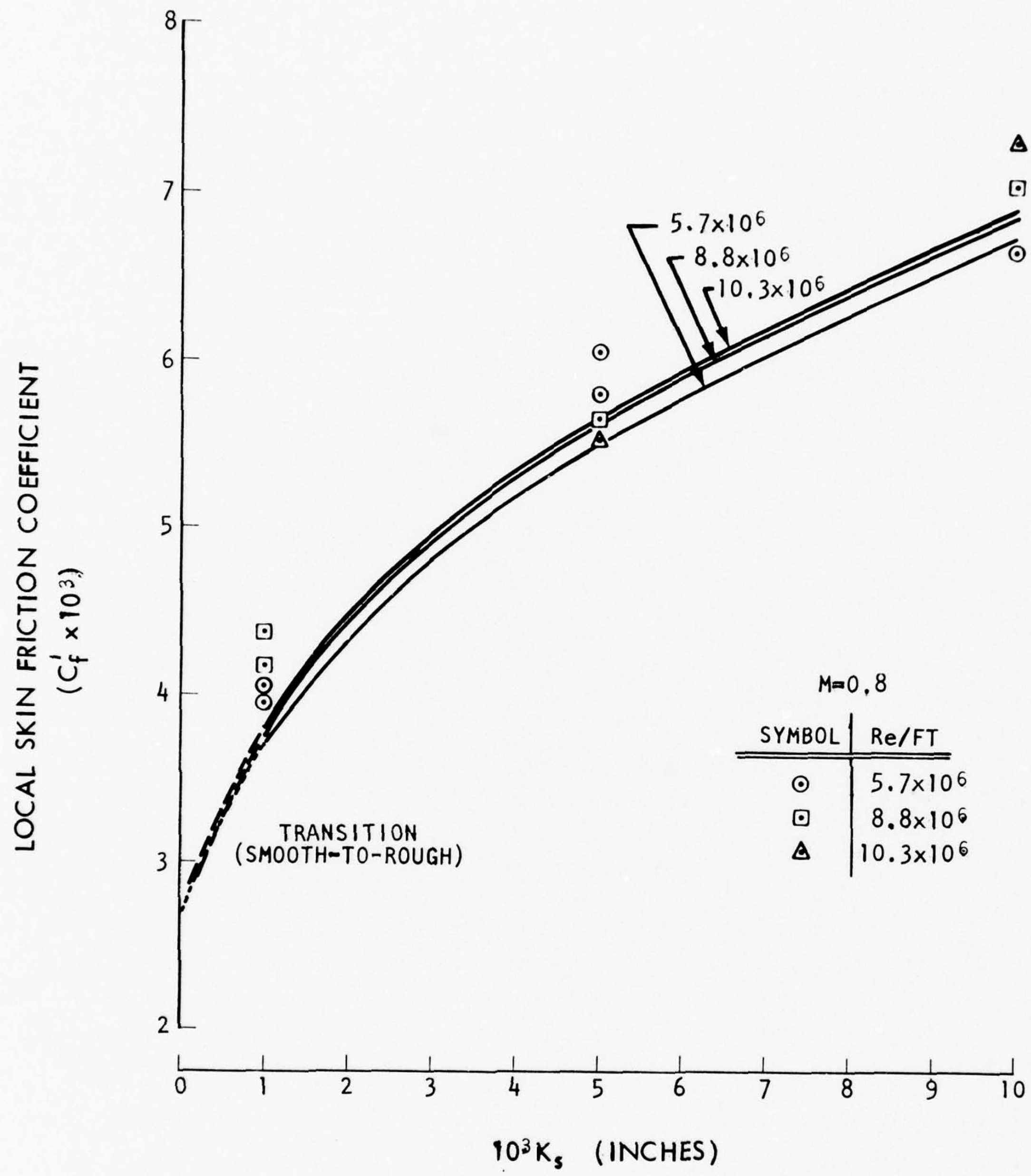


Figure 4-14 Effect of roughness on incompressible skin friction

4.6.4 Sandpaper Grit Sizes

Several different grit numbers of sandpaper were used in the tests and one of the problems encountered was obtaining a consistent roughness measurement. The grit numbers commonly used to classify sandpaper refer to the number of mesh openings per inch in the sieves used to sort the particles. The nominal size of the opening is given below.

Sieve number	Nominal opening diameter
100 meshes/in	.0059 in.
220	.0029 in.
600	.0011 in.

However, most abrasive particles used in modern sandpaper are not of uniform shape, but tend to be elongated, with a fineness ratio up to 2. Further, as the sandpaper is made, the particles are electrostatically set more or less on end as they are glued to the backing material, so that the nominal diameter listed above is not a measure of the profile roughness. The sandpaper used for the reference samples was obtained from a variety of manufacturers and one sample was made by glueing carborundum particles directly onto the metal surface of a test specimen. The particle sizes were observed in microscopes in an attempt to characterize the particles. In most cases there was a large variation in particle size with occasional large particles being up to 3 times greater in their longest dimension than the average particle. The coarse sandpapers had more elongated particles with the 600 grit particles being most "spherical" - or at least their height and diameter were the same dimension. Due to the importance of the grit size to the validation of the correlating curves, several independent measurements were made of the particles. The first was made using a calibrated eyepiece at the test bench in the wind tunnel. Later the samples were observed in the test lab using a microscope and finally the Particle Identification group in Quality Assurance made measurements using differential interference micros copy. This final report is given in Appendix IV. In addition one of the sandpaper manufactures sent profile measurements of 100, 150, and 220 grit specimens. Unfortunately the probe radius used for these profiles was significantly

larger (.015 and .011) than most of the particles, and the maximum troughs on the profiles had to be confirmed from the data. In all cases the largest particles over a space of about 5 particles were noted, for it was assumed that the flow would be governed by the larger particles.

The results are given in Table 4-1 and show the following results:

Grit Size	Nominal	Measured height
100	.006 in.	.01 in.
150	.004	.007
220	.003	.005
600	.001	.001

The measured values were used to plot the measured drag (or skin friction coefficient) on Figures 4-13 and 4-14 above and show very good correlation.

TABLE 4-1 GRIT SIZE FROM VARIOUS MEASUREMENTS
(ALL DIMENSIONS IN 10^{-3} IN)

MEASUREMENT		GRIT SIZE		
	100 (6 mils)	150 (4 mils)	220 (3 mils)	600 (1 mil)
Test Bench	length 10 to 12		5	
	Fineness Ratio 2		2	
Lab	Height 7 to 15		5	.7 to 1
	Fineness Ratio 2		1 to 1.5	1
Particle Lab (Coated Samples)	Height 9 to 10	7	5	1+
	Fineness Ratio 1	1-	1-	1+
Manufacturer	Height 11	8	8	
	Diameter -			

5.0 TEST RESULTS

Groups of thermally damaged specimens were chosen to be typical of various skins and coatings and then drag tested at $M = .5$ and .8 for various Reynolds numbers. The drag was converted to a local skin friction coefficient and the equivalent sand roughness obtained from Figures 4-13 and 4-14. The results were than plotted against absorbed heat and are presented as Figures 5-1 and 5-2 (for aluminum skins) and 5-3 thru 5-6 (for fiberglass). Due to the difficulty in testing and interpreting the results for debonded, delaminated or burnt through skins most of these results are only for specimens with paint damaged or removed. However some of the extremely damaged specimens were tested and are discussed in Section 5.3. Appendix 3 gives a detailed history of run conditions.

The figures show that after the initial increase in drag due to paint blistering, there is no significant increase up to paint removal. This stage occurred at an absorbed heat level of about 80 cals/cm^2 for most specimens.

The equivalent sand roughness was obtained from individual plots of C_f' vs k_s for each test mach number and Reynolds' number, although to the accuracy of the test, probably one curve would have been sufficient. The skin friction coefficient plots Figures 5-1 and 5-3 show a much greater consistency than do the k_s plots. This is due, of course, to the low slope on the curves used to obtain k_s from C_f', so that a change of 25% is C_f' translates to a change of about 100% in k_s in much of the region of interest.

5.1 ALUMINUM SKIN SPECIMENS

Specimens of skin thickness 12, 20, 25, 32, 40, 50 and 63 mils were tested for their skin friction drag. They had generally been exposed at thermal levels of 30 to 50 cals/cm^2. Beyond this level there was generally debonding of the face skin with a separated edge or significant bulge in the skin, which made it impossible to obtain meaningful drag results in the test set-up. Details of these runs are given in Appendix 3, and the results are plotted in Figures 5-1 and 5-2. Between 5 and 15 runs were made on each specimen, and the resulting values of skin friction coefficient and equivalent sand roughness averaged to give the data points in these figures. Surface profile measurements were taken

SYMBOL	SKIN			COATING			SAMPLE NUMBER
	MAT'L	THICK	CURE TEMP	MAT'L	COLOR	THICK	
○	AL	12	250	ENAM	BLK	3	1
△		20					3
□		25					4
◇		32					5
▽		32	350				10
▷		40					25
◁		50					26
◊		63					27
×	UNPOLISHED PLATE						
+	POLISHED PLATE						

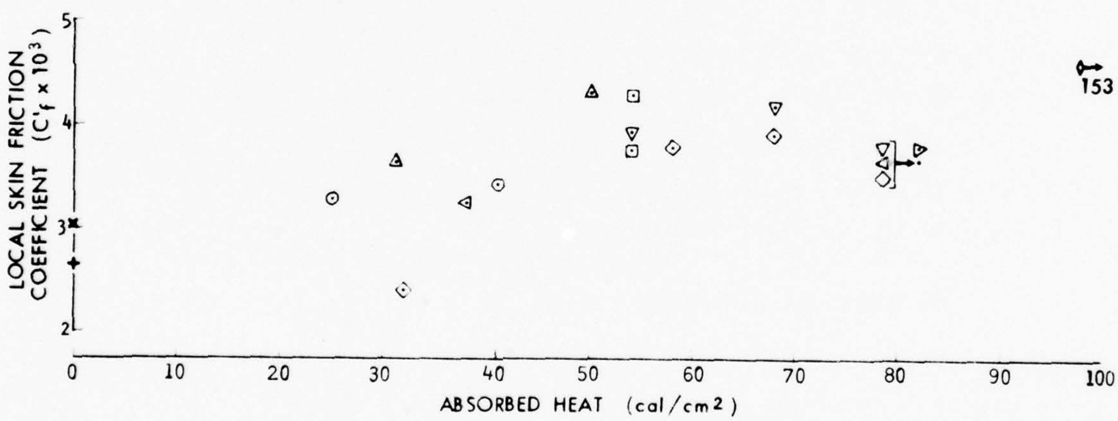


Figure 5-1. Absorbed heat versus local skin friction coefficient for aluminum skin samples.

SYMBOL	SKIN			COATING			SAMPLE NUMBER
	MAT'L	THICK	CURE TEMP	MAT'L	COLOR	THICK	
○	AL	12	250	ENAM	BLK	3	1
△		20					3
□		25					4
◇		32					5
▽		32	350				10
▷		40					25
◁		50					26
◊		63					27
×	UNPOLSIHED PLATE						
+	POLISHED PLATE						

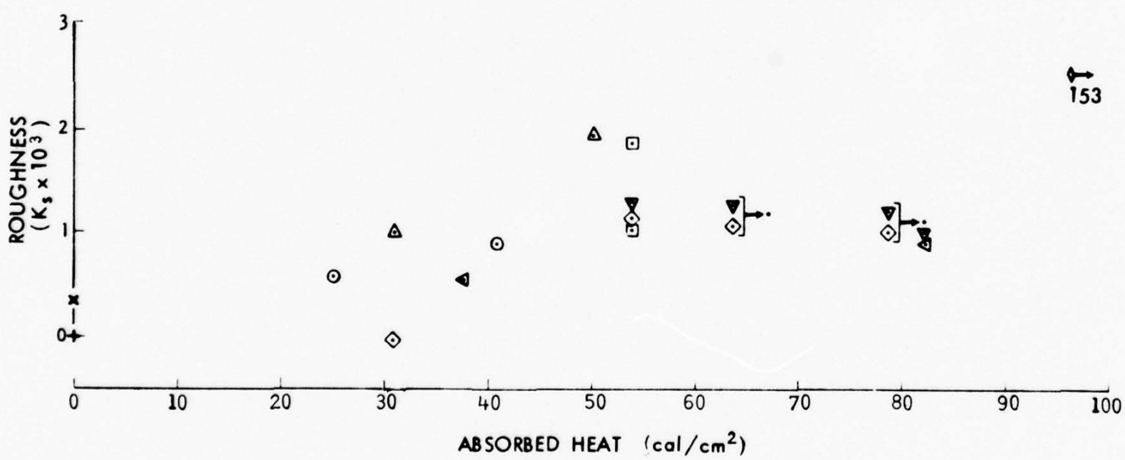


Figure 5-2. Equivalent sand roughness for aluminum skin samples.

	SKIN			COATING			SAMPLE NUMBER
	MAT'L	THICK	CURE TEMP	MAT'L	COLOR	THICK	
○	EPX/ E-FGL	13.5	350	ENAM	BLK	4	11
□		20					12
◇		23					13
△		13.5	250				14
◊		20					15
▽		30					16
×	UNPOLISHED PLATE						
+	POLISHED PLATE						

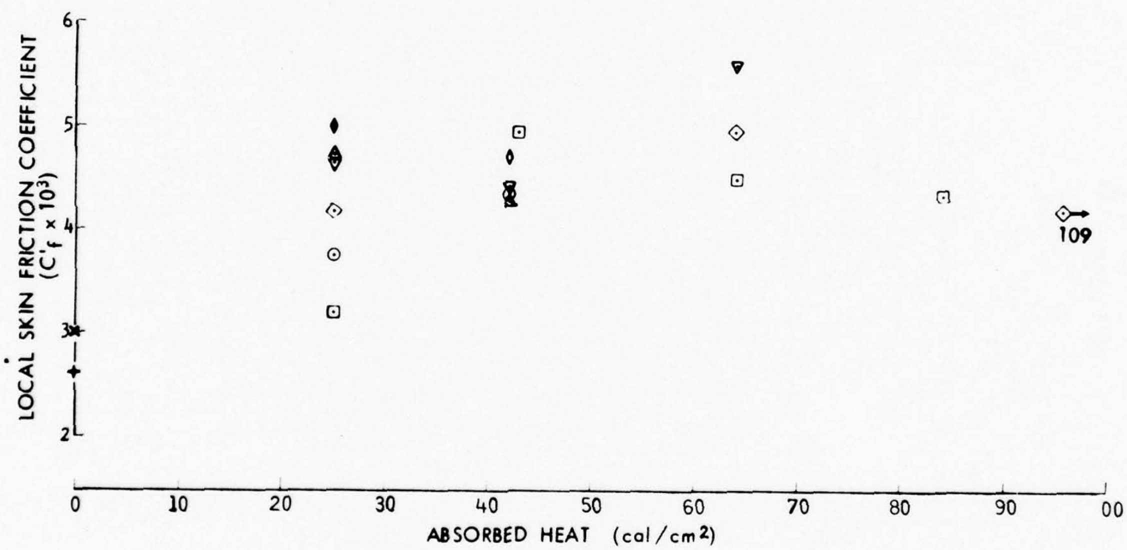


Figure 5-3. Absorbed heat versus local skin friction coefficient for fiberglass/enamel samples.

	SKIN			COATING			SAMPLE NUMBER
	MAT'L	THICK	CURE TEMP	MAT'L	COLOR	THICK	
○	EPX/ E-FGL	13.5	350	ENAM	BLK	4	11
□		20					12
◇		23					13
△		13.5	250				14
◊		20					15
▽		30					16
×	UNPOLISHED PLATE						
+	POLISHED PLATE						

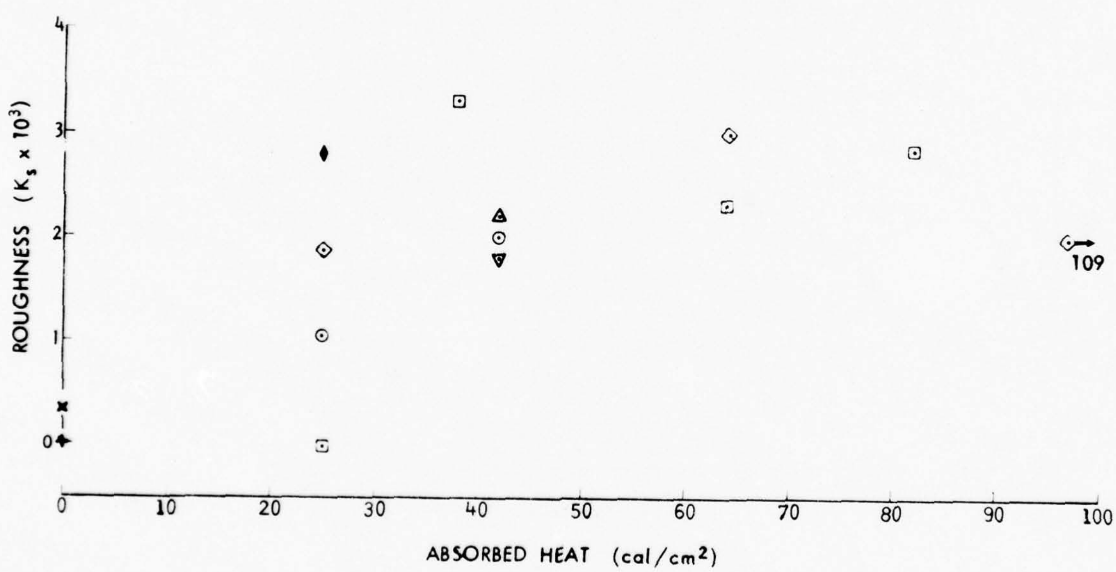


Figure 5-4 Absorbed heat versus equivalent sand roughness for fiberglass/enamel skin samples.

SYMBOL	SKIN			COATING			SAMPLE NUMBER
	MAT'L	THICK	CURE TEMP	MAT'L	COLOR	THICK	
○	EPX/E-FGL	30	250	ASTRO	BLK/WHT	16	17
△	↓	↓	↓	↓	BLACK	15	18
□					WHITE	18	19
◇				FLCBM	BLK/WHT	13	20
▽				↓	WHITE	13	21
▷				LACQUER	BLACK	5	22
◁	↓	↓	↓	EN/PSUL	↓	↓	23
◆	EPX/S-FGL	50	350	ENAMEL			24
★	EPX/GR	20	250	↓	↓	↓	29
×	UNPOLISHED ALUMINUM PLATE						
+	POLISHED ALUMINUM PLATE						

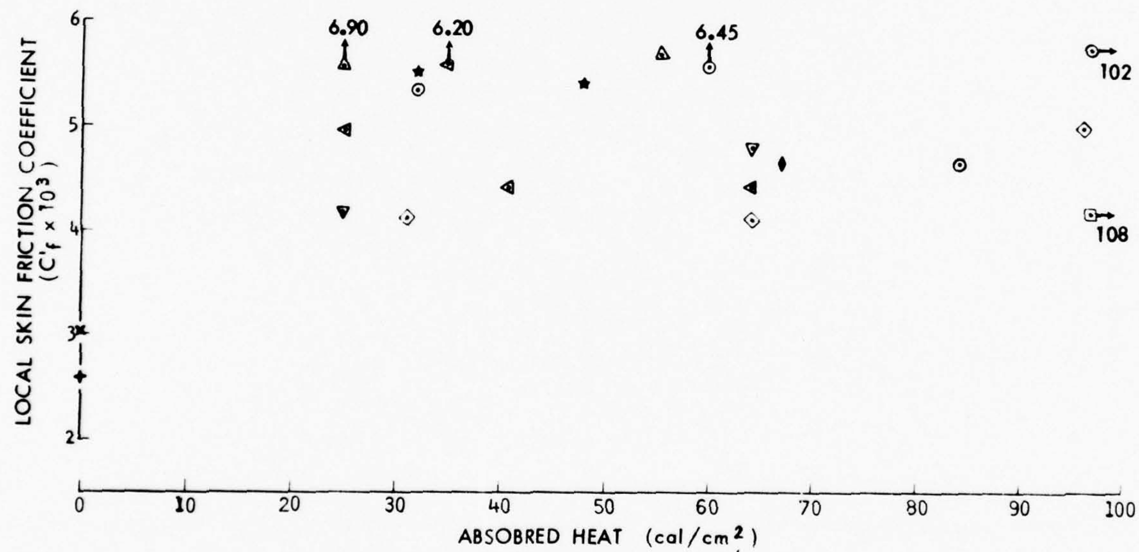


Figure 5-5. Absorbed heat versus local skin friction coefficient for fiberglass/thick coating and graphite skin samples.

SYMBOL	SKIN			COATING			SAMPLE NUMBER
	MAT'L	THICK	CURE TEMP	MAT'L	COLOR	THICK	
○	EPX/E-FGL	30	250	ASTRO	BLK/WHT	16	17
△					BLACK	15	18
□				↓	WHITE	18	19
◇				FLCBM	BLK/WHT	13	20
▽				↓	WHITE	13	21
▷				LACQUER	BLACK	5	22
◁	↓	↓	↓	EN/PSUL			23
◆	EPX/S-FGL	50	350	ENAMEL			24
★	EPX/GR	20	250	↓	↓	↓	29
×	UNPOLISHED ALUMINUM PLATE						
+	POLISHED ALUMINUM PLATE						

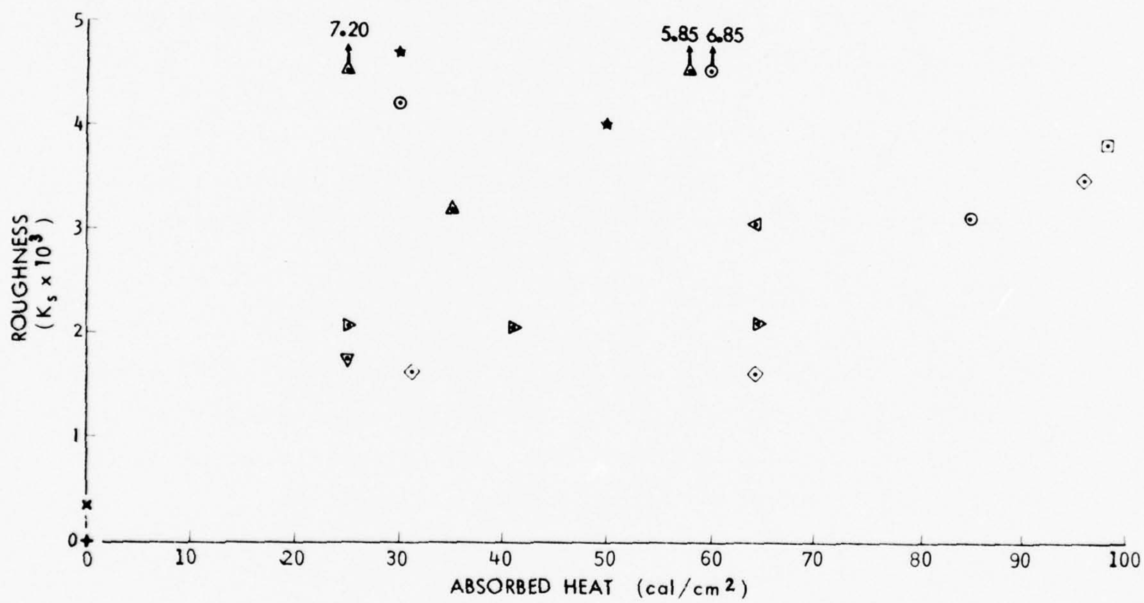


Figure 5-6. Absorbed heat versus equivalent sand roughness for fiberglass/thick coatings and graphite.

on a group of specimens (Specimens 10, 32 mil aluminum with black enamel) to determine whether there was a direct correlation between physical roughness and equivalent sand roughness. A typical plot is shown as figure and the remainder are in Appendix 4. The measurements were made along the centerline, at the forward and aft thirds of the plate. The results are (with some subjective judgement on choosing extreme values):

Specimen	Fluence	k_s, eq	Max. height (profile)
10-1	30 cals/cm^2	*Smooth	.5 mil
10-2	52	1.3 mil	1.0 to 1.5
10-3	64	1.2	1.0 to 1.2
10-4	79	1.0	1.0 to 1.2

*10-1 gave suspicious results. 5-2 was identical in appearance and gave "smooth" values.

It could thus appear that a direct profile reading could be used to give a reasonable accurate equivalent sand roughness measurement for the aluminum samples. However, this did not appear to hold true for the fiberglass.

5.2 FIBERGLASS SKIN SPECIMENS

For the fiberglass specimens, skin thicknesses of 13.5, 20 and 30 mils were tested. The coating was most often black enamel, but other coatings were used. Thermal exposure levels ranged somewhat higher than for the aluminum, with the maximum being about 80 cals/cm^2 before debonding occurred.

Fairly consistent results were obtained for the specimens with a coating of black enamel (Specimens 11-16) and also for the low thermal fluence levels for the more exotic thick coatings. However, as these thicker coatings flowed, blistered, and peeled off in thick layers, the drag results gave a much greater data scatter (Specimens 17 - 19). Further, the graphite/epoxy samples (29) gave the highest drag results for a simple black enamel coating. Examination of the samples showed a slightly different blistering than observed for the fiberglass/enamel specimens.

Surface profiles measurements were taken for several of the fiberglass specimens (and also the graphite specimen) and the results are summarized below. The profile traces may be found in Appendix 4.

Specimen	Fluence	k_s, eg.	Max ht. (Profile)
11-3	24 $cals/cm^2$	1 mil	3 mils
12-2	24	.5	.5
12-3	37	3.4	1 to 1.5
12-4	62	2.4	2
12-7	80	1.9	2 to 3
14-1	24	2.8	4 to 5
29-1	30	3.7 to 6.6	1 to 5
29-2	47	4.6	2 to 3

From these results it would seem that the correspondence that occurred for the aluminum specimens was fortuitous, and that no simple relationship exists between surface profiles and equivalent sand roughness for thermally damaged specimens.

These fiberglass skin specimens gave a higher drag and thus roughness measurement than did the aluminum ones. The increase in drag over the smooth plate was nearly 50% greater for fiberglass than aluminum, and this translated into a k_s value of about 2.5×10^{-3} inches compared to 1.25×10^{3} inches for the aluminum. Although there is considerable scatter in the roughness values shown in figure 5-4 there does not appear to be a significant trend associated with skin thickness or bonding cure temperature.

The graphite/enamel specimens gave readings well on the high side of the fiberglass/enamel specimens and should be considered separately when estimating drag.

The fiberglass specimens with a thick coating (17, 18 and 19) all gave much higher readings for low levels of absorbed heat (20 - 60 $cals/cm^2$). At this level the paint flowed in thick layers and formed very lumpy surfaces. At higher levels of absorbed heat, the paint was burnt or flowed off the specimens and their drag values came down to the usual fiberglass skin levels.

5.3 SEVERELY DAMAGED SURFACES

In addition to the specimens described above, some drag measurements were made on specimens where there was severe skin damage. Only a limited time was spent on these measurements for it did not seem that any generalizations could be made from the data. However the condition of the specimens was interesting and tests were made. The specimens were:

17-7 E-Fiberglass, 30 mils skin, 15 mil coating of Astrocoat, black over white. About 100 cals/cm^2.

23-5 E-Fiberglass, 30 mils skin, 1-3 mils Black polysulphide, 78 cals/cm^2.

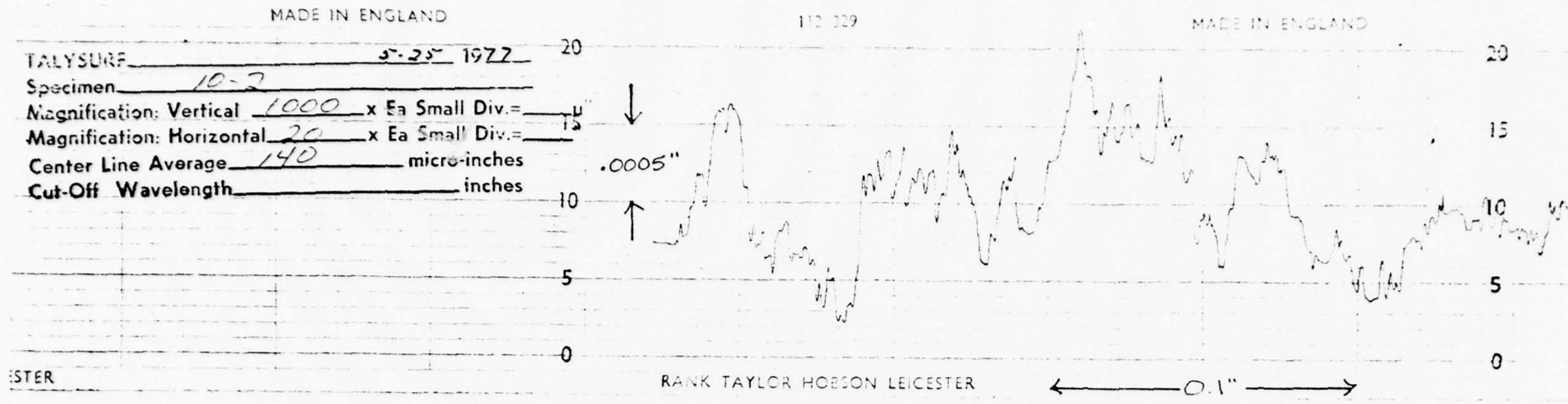
26-6 Aluminum, 50 mils, 3 mils black enamel 145 cals/cm^2. Completely exposed honeycomb.

The two fiberglass specimens had one layer of fiberglass completely burnt through in the center (about 2" in dia.) and numerous stray ends of fiberglass. They were debonded and bulged in the center, but had good attachment of the skin all along the leading edge. Specimen 23-5 initially had fiberglass remaining along the leading edge and both sides, whereas 17-7 had some fibers attached across the training edge as well. During the runs all the top layer of fiberglass, except for a 1" strip at the leading edge, gradually was removed.

The "aluminum" specimen lost its face sheet in the thermal exposure tests, but the specimen was flat, and was able to be mounted flush with the tunnel floor. The exposed edges of the honeycomb generally had a layer of epoxy remaining.

The values for the skin friction coefficient are given below (full run conditions are listed in Appendix 3).

The fiberglass samples in their final condition (i.e. very few "strings") could be considered equivalent to a sand roughness of .010". However, the debonding that occurred makes any extrapolation of these results to any real aircraft surface very dubious. The same comment really applies to the exposed honeycomb specimen - once one surface of a honeycomb structure is lost, the remaining strength is minimal or the structure would probably be completely lost in flight. However we did go through the calculation procedure to obtain a roughness value, and obtained a $k_s = 0.1"$ with $x/k_s = 35$.



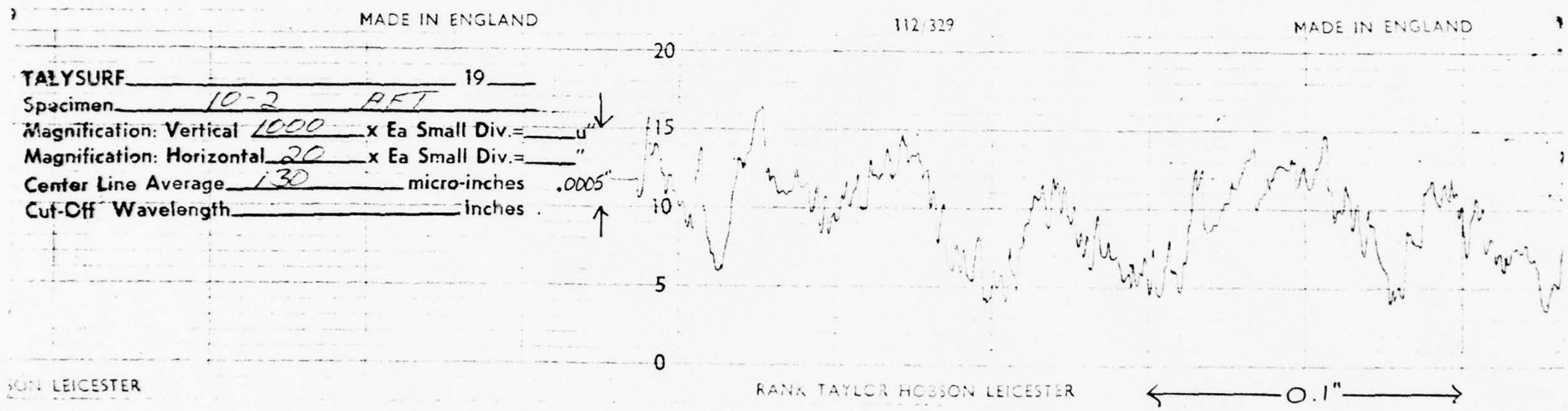


FIGURE 5-7 PROFILE OF EXPOSED ALUMINUM/ENAMEL SURFACE. SPECIMEN 10-2

TABLE 5-1

VALUES OF LOCAL SKIN FRICTION COEFFICIENT

Specimen No.	Mach. No.	$10^3\ C'_{fi}$
17-7	.5	23.9
	.5	20.7
	.5	14.9
	.5	15.
	.5	16.
	.5	7.6
23-5	.5	6.8
	.5	7.2
	.8	8.5
	.8	8.0
26-6	.5	14.1
	.5	14.3
	.8	15.6
	.8	17

6.0 CONCLUSIONS AND RECOMMENDATIONS

Typical thin skin samples of aircraft structure were exposed to a simulated nuclear thermal pulse and then tested for increased drag due to surface damage.

The test and analysis techniques used for this program were adequate to give good final drag results, and to translate the drag into equivalent sand roughness. Coating damage began to occur at about 10 cals/cm^2 of absorbed heat, and continued until much of the coating was burnt off at about 80 - 100 cals/cm^2. For the thin aluminum skins, significant skin damage occurred at about 60 cals/cm^2, with melting and debonding. For the thicker aluminum skins there was significant debonding at 80 - 100 cals/cm^2. The fiberglass skins debonded at about the same level of absorbed heat, and subsequently the surface layer or layers of glass cloth disintegrated.

The equivalent sand roughness for the aluminum specimens was generally lower (for the same heat absorbed) than for the fiberglass ones,about .001" compared to .002". In both cases there was a quick increase in roughness at about 20 cals/cm^2 and then the roughness remained roughly constant up to the stage of skin damage (debonding, etc.).

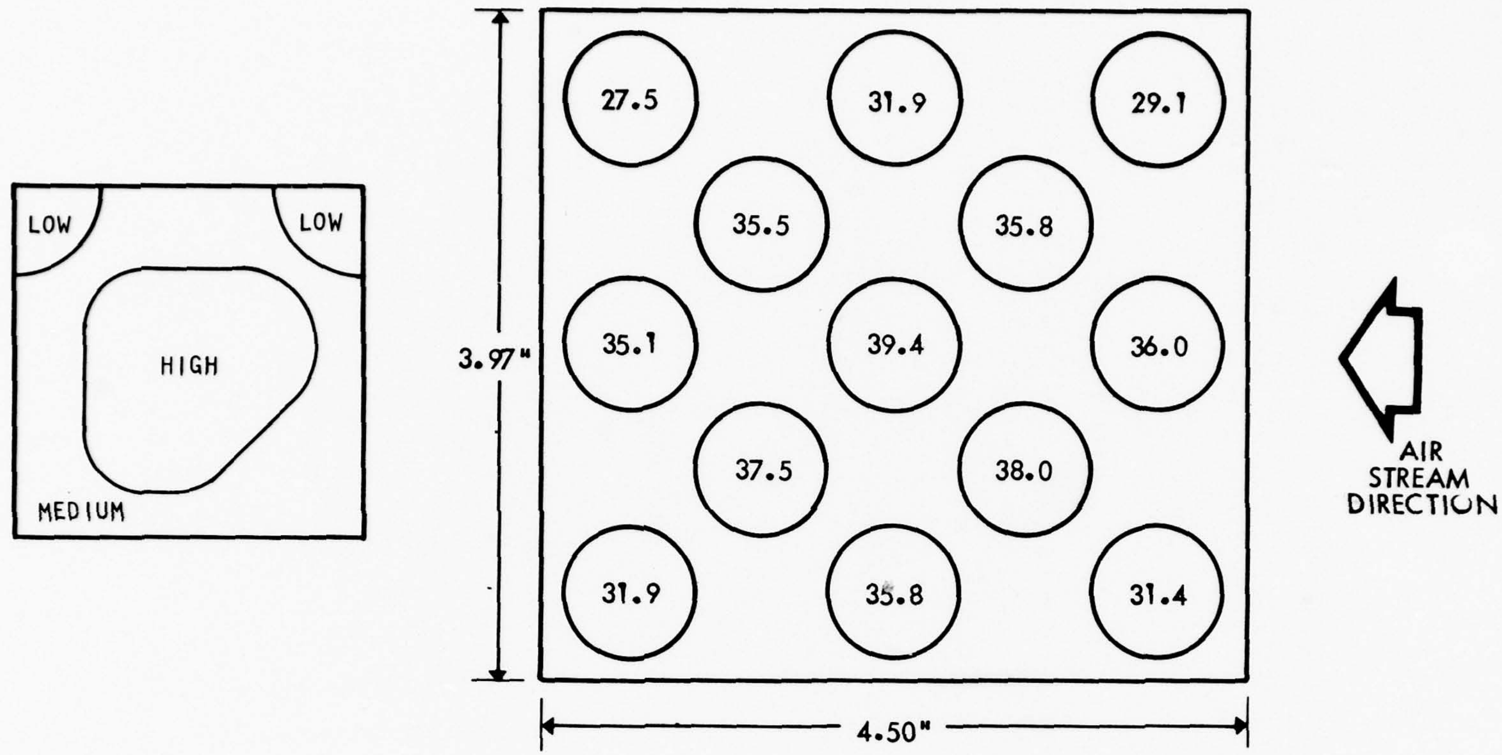
An attempt was made to correlate the drag derived "sand roughness" with profile measurements of the damaged surfaces, but no consistent results were obtained.

The current work gives sufficiently accurate data for further skin friction drag calculations to be made on actual aircraft. However, these tests did not address the problem of profile drag resulting from melted and debonded skins. Further it is not clear for large areas of skin (for example, a complete aileron) whether debonding would occur uniformly or not so that the residual strength of these honeycomb structures with severe skin damage is still not known.

APPENDIX 1

UNIFORMITY OF IRRADIATION

Examination of exposed specimens indicated that there might be some non-uniformity of irradiation over the specimen face. Because of this concern, peak flux readings were taken at thirteen different locations over the specimen area. Figures I-I through I-V are "maps" showing peak flux vs. location for five different levels of irradiation. Flux meter size and locations are to scale. The center of the specimen face always has the highest flux, with lower, but not necessarily uniformly lower, fluxes toward the specimen edges. Toward the upper flux limit the center area peak flux is approximately 30% higher than the lowest reading near the edge of the specimen. At lower peak fluxes the difference between the center (high) reading and the lowest edge reading decreases to a minimum of 15%. We have been advised that there is some concern that the absolute value of these measurements may not be correct, but a final determination of the correct flux measurement has not been made.



DATE: 4-6-77

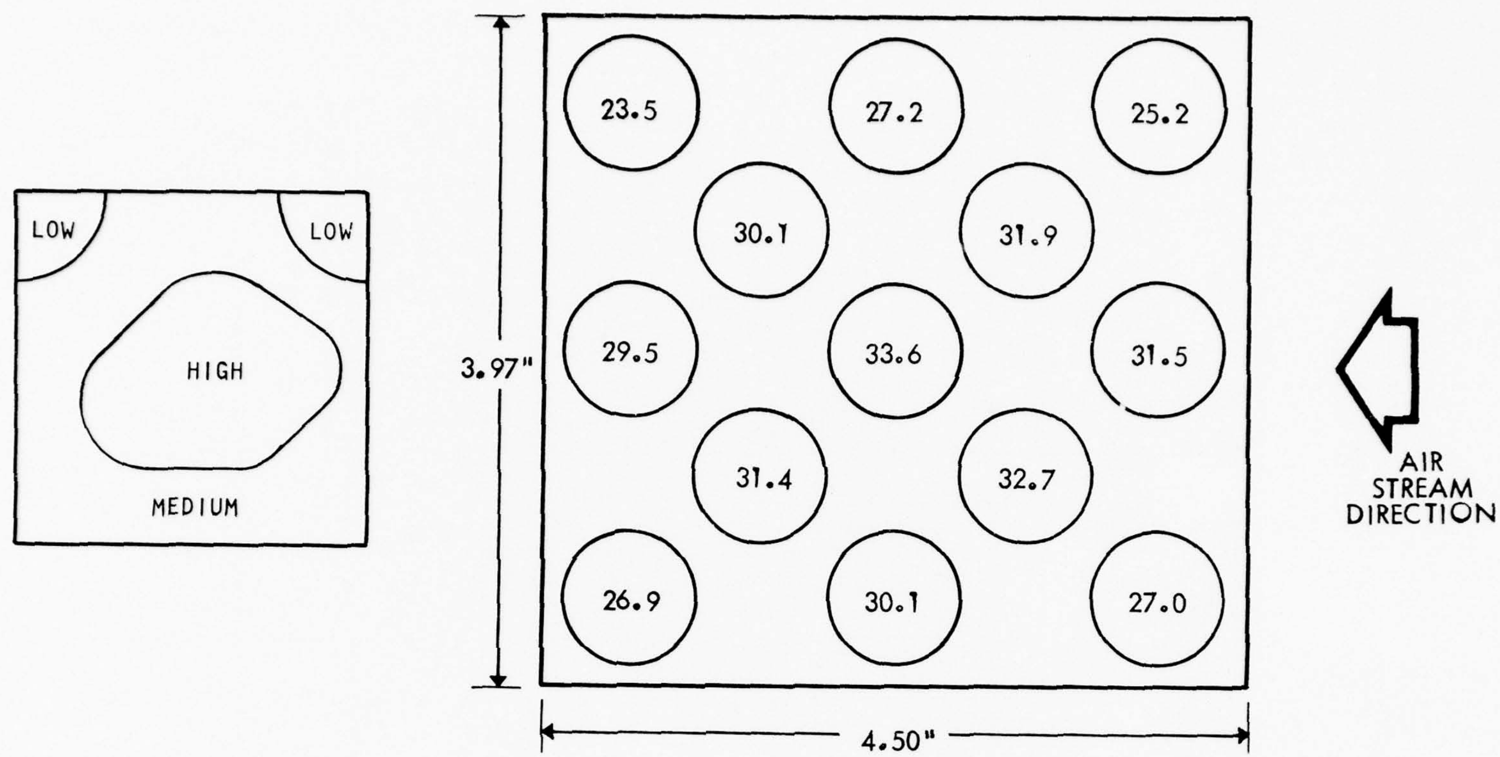
CENTER PEAK: 39.4 cal/cm^2-sec

AVERAGE PEAK: 33.5 cal/cm^2-sec

DISTANCE: 3"

MAXIMUM VARIATION: 30%
(CENTER = 100%)

Figure A1-1 Cal/cm^2 - sec peak after four second exposure looking at the sample surface (Face).



DATE: 4-5-77

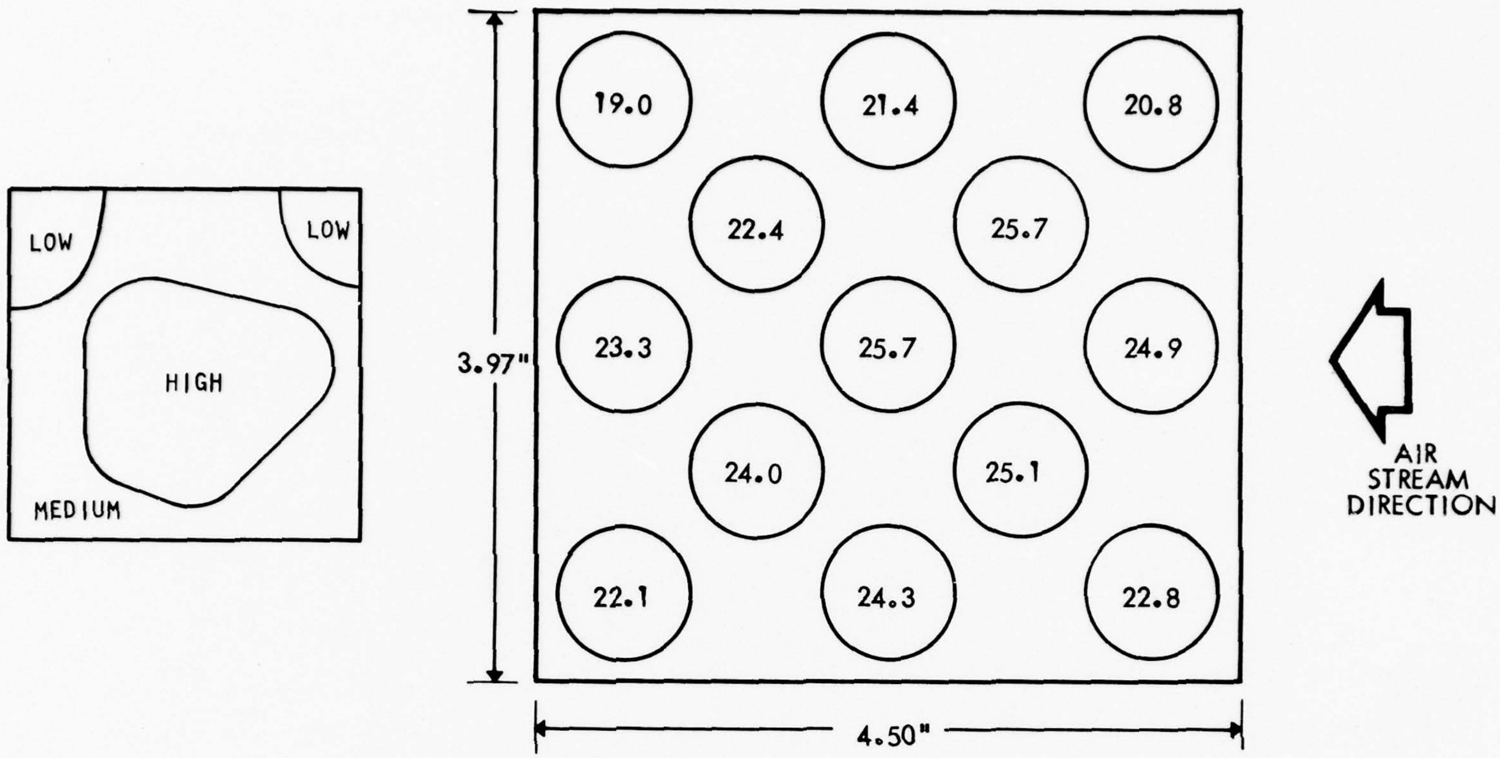
CENTER PEAK: 33.6 cal/cm^2-sec

AVERAGE PEAK: 28.5 cal/cm^2-sec

DISTANCE: 4"

MAXIMUM VARIATION: 30%
(CENTER = 100%)

Figure A1-2 Cal/cm^2-sec peak after four second exposure looking at the sample surface (Face).



DATE: 4-5-77

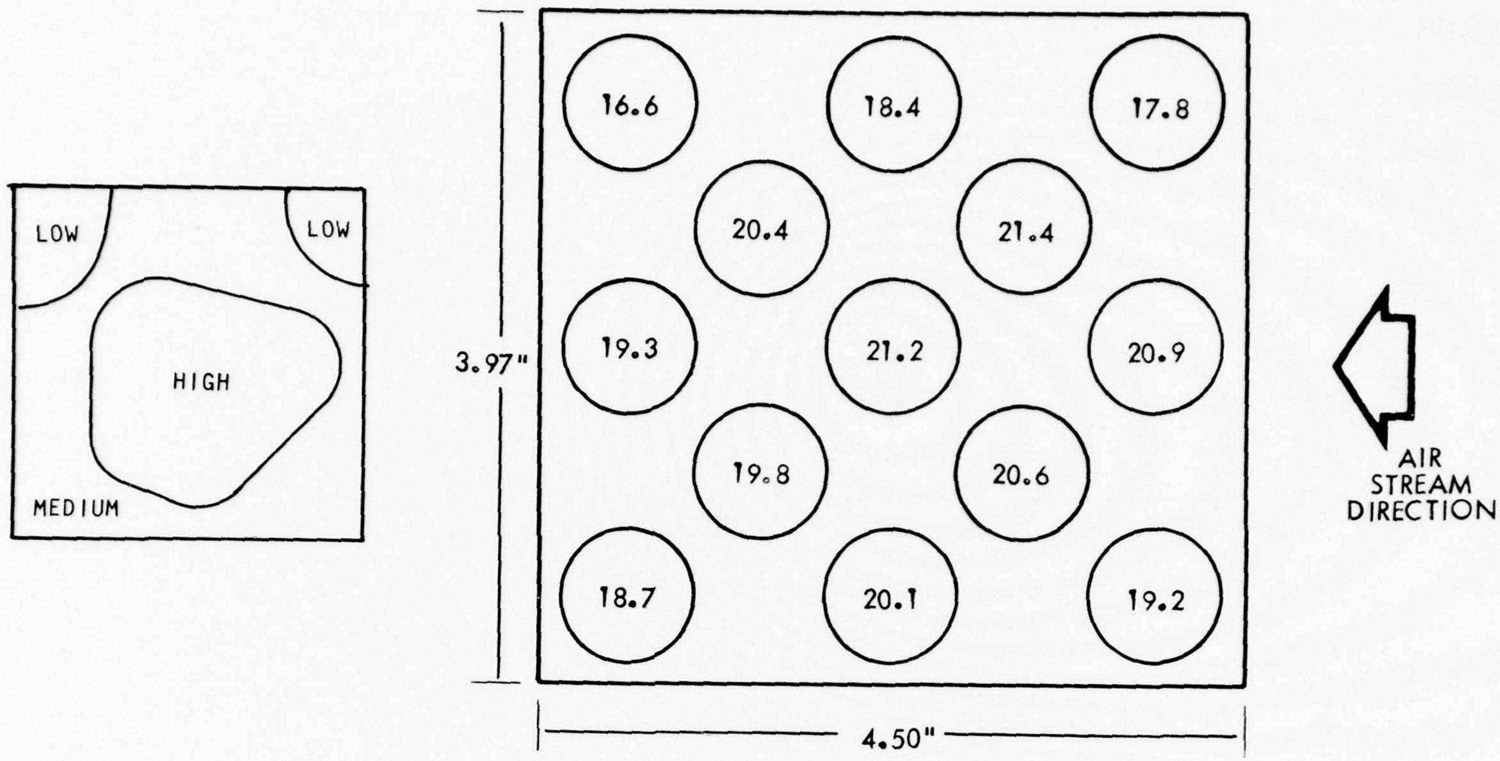
CENTER PEAK : 25.7 cal/cm²-sec

AVERAGE PEAK: 22.3 cal/cm²-sec

DISTANCE: 5"

MAXIMUM VARIATION: 26%
(CENTER = 100%)

Figure A1-3 Cal/cm^2 - sec peak after four second exposure looking at the sample surafce (Face).



DATE: 4-5-77

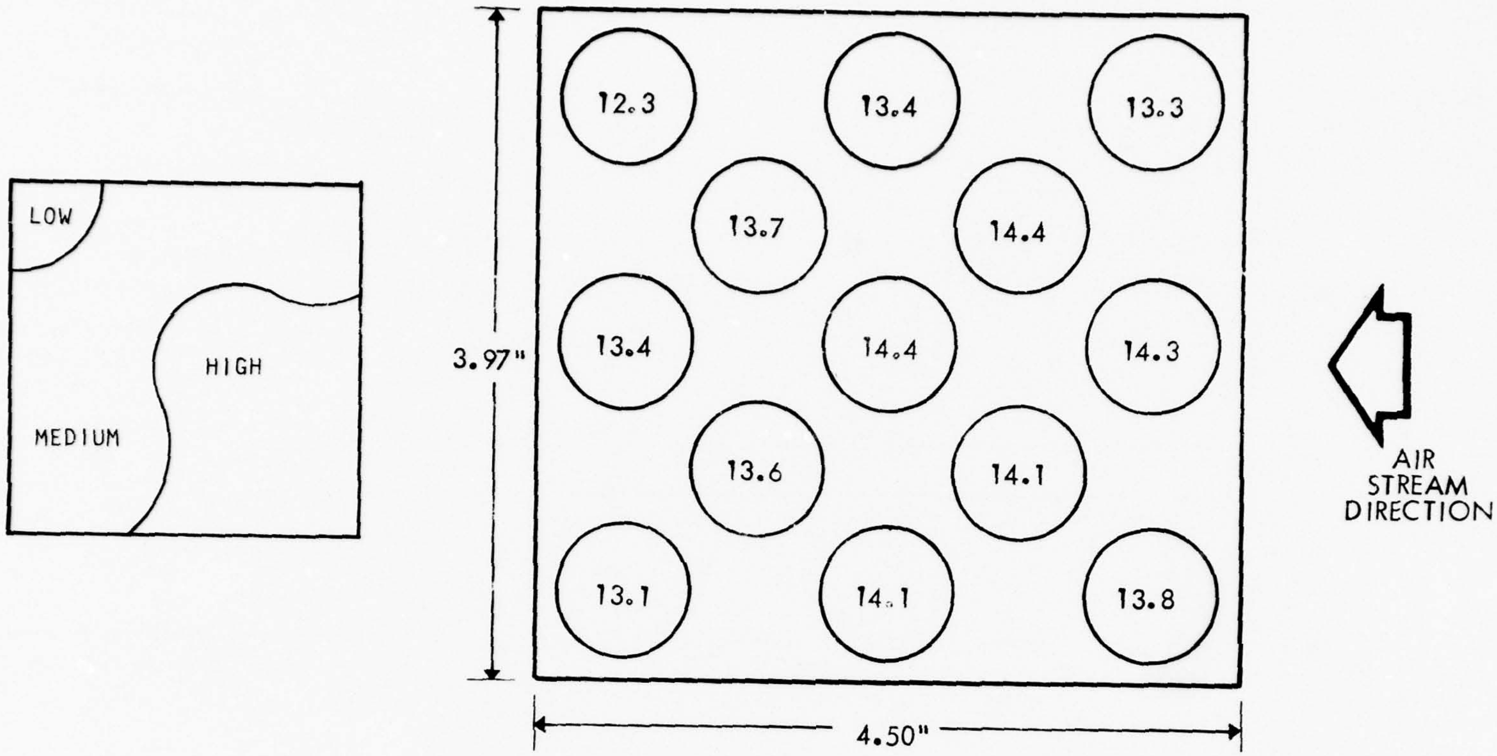
CENTER PEAK: 21.2 cal/cm^2-sec

AVERAGE PEAK: 19.0 cal/cm^2-sec

DISTANCE: 6"

MAXIMUM VARIATION: 22%
(CENTER = 100%)

Figure A1-4 Cal/cm^2-sec peak after four second exposure looking at the sample surface (Face).



DATE: 4-5-77

CENTER PEAK: 14.4 cal/cm^2-sec

AVERAGE PEAK: 13.3 cal/cm^2-sec

DISTANCE: 8"

MAXIMUM VARIATION: 15%
(CENTER = 100%)

Figure A1-5 Cal/cm^2-sec peak after four second exposure looking at the sample surface (Face).

APPENDIX 2

RESPONSE OF SPECIMENS TO THERMAL EXPOSURES

This appendix consists of data sheets listing each thermal exposure and specimen response. Specimens are listed by series, i.e. 1-X, 2-X, etc. and within each series in order of increasing thermal exposure fluence.

An estimate is given of thermal absorptivity of the specimen surface after the thermal exposure. This was done with the thought that it will help in estimating total absorbed energy. Peak flux/pulse duration are shown since pulse shape has an effect on response of some materials. This data is shown in sequence as cal/cm^2-sec/seconds, e.g. "20/1.0" would indicate a peak flux of 20 cal/cm^2-sec with 1.0 seconds to reach the peak flux. Total "power on" (to the lamps) time is typically 0.25 seconds longer than the time to peak flux, i.e. it requires approximately 0.25 seconds of power on before there is a measurable output from the lamps. The data sheets also show a total fluence for each exposure. This was obtained by a mechanical planimeter measurement of area under the flux meter curve, including lamp cool down. The quartz lamps were run at constant voltage for all exposures. Flux and fluence were varied/controlled by lamp distance from the specimens and by pulse length. A flux meter curve was obtained each time lamp distance and/or pulse time were changed. It was then assumed that flux and fluence remained constant for subsequent runs so long as the settings were not changed (for at least 20 to 30 runs). Equipment check out has shown that such repetitive runs are very constant.

Except where otherwise shown each specimen was exposed to one, only, thermal pulse.

The following notes, etc. apply throughout the data sheet thermal response descriptions:

a. " % of specimen area" is frequently used to quantify results. The percentage is a rough estimate, based on visual examination, with 100% being the entire face of the specimen.

b. Unless otherwise noted it can be assumed that damage is reasonably progressive as thermal fluence increases within any one specimen series. For example, when "paint is completely removed from 50% of specimen area," it should be assumed that the other 50% has paint remaining and that the condition of the remaining paint is comparable to the paint damage description for lower fluence exposures of that series of specimen.

c. Fiberglass substrates typically delaminate one ply at a time, however, in most cases this could not be determined without further damage to the specimen. Accordingly, most such data is merely reported as delamination without indicating which plies are involved.

d. When the fiberglass plys "disintegrated", the fibers parallel to the air stream tended to break. Some of the perpendicular fibers frequently remained attached at one end (near one side of the specimen) with the other end drawn downstream. This formed a "tail" which always produced a loud, distinctive sound in the wind tunnel.

e. Aluminum skin delamination always occurred at the skin/honeycomb core interface and always involved some warping of the aluminum skin. Unless otherwise stated, it should be assumed that aluminum skin/core delamination starts along the downstream edge of the specimens and progresses more or less uniformly toward the upstream edge of the specimens.

f. Descriptions of paint damage frequently include a comment regarding exposed primer. It appears that the black enamel topcoat typically forms small blisters with separation occurring at the enamel/primer interface. The blistered enamel tends to rapidly ablate (having lost the cooling effect of the substrate heat sink) exposing the yellow, more reflective, primer within the small blister area. When percentages of exposed primer are given, these are rough estimates of the total area of visible primer as a percent or the total area of the specimen face.

g. Epoxy-fiberglass laminate is usually referred to as "FRP," i.e. fiber reinforced plastic.

Table A2-1

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
1-1	Long pulse - specimen discarded	--	--	--
1-2	Paint lost gloss over most of specimen area	.97	20/1.0	25
1-3	Paint slightly blistered	.97	32.8/1.0	42
1-4	Paint blistered; some bare aluminum. Skin delaminated for one inch, center of downstream edge	.90	30/1.5	51
1-5	Same as 1-4	.90	30/1.5	49
1-6	Paint similar to 1-4. Skin delamination extends 1.5 inches, center of downstream edge.	.95	32.7/1.5	54
1-7	Short pulse - specimen discarded	--	--	--
1-8	Paint completely removed in 50% of specimen area. Skin delaminated in 20% of specimen area.	.70	30/2.0	64
1-9	Paint completely removed in 95% of specimen area. Small area of skin disintegrated.	.70	30/2.5	82

Table A2-2

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
2-1	Paint lost gloss over most of specimen area - still smooth.	.97	25/1.0	31
2-2	Primer showing in 10% of specimen area. Skin delaminated; 1.5 inches center of downstream edge.	.95	30/1.5	49
2-3	Paint blistered. Skin delaminated for 2.0 inches, center of downstream edge.	.95	32.7/1.5	54
2-4	Paint completely removed from 15% of area. Skin delaminated 20% of area.	.85	30/2.0	64
2-5	Paint completely removed from 75% of specimen area. Skin delaminated, 15% of specimen area.	.75	30/2.5	82
2-6	Paint completely removed from 95% of specimen area. Skin delaminated, 20% of specimen area.	.70	30/2.0	96
2-7	Paint completely removed from 95% of specimen area. Small area of skin disintegrated/tore out. 15% of skin area delaminated.	.70	30/3.5	109

Table A2-3

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
3-1	Paint lost gloss over most of specimen area. Paint is still smooth.	.97	25/1.0	31
3-2	Paint blistered, primer showing in 5% of specimen area.	.95	30/1.5	49
3-3	Paint blistered. Skin delaminated for 2 inches, center - downstream edge of specimen.	.95	32.7/1.5	54
3-4	Paint completely removed from 5% of specimen area. Skin delaminated in 30% of specimen area.	.90	30/2.0	64
3-5	Paint completely removed from 20% of specimen area. Skin delaminated in 30% of specimen area.	.85	30/2.5	82
3-6	Paint completely removed from 95% of specimen area. Skin delaminated in 40% of specimen area.	.75	30/2.0	96
3-7	Paint completely removed from 95% of specimen area. Skin delaminated in 40% of specimen area.	.75	30/3.5	109
3-8	Paint completely removed from 95% of specimen area. Skin delaminated, 30% of specimen area. Skin disintegrated, tore out, 15% of area.	.80	30/4.0	124

Table A2-4

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
4-1	Paint lost gloss over most of specimen area - still smooth.	.97	25/10	31
4-2	Paint blistered, primer showing in 5% of specimen area.	.95	32.7/1.5	54
4-3	Same as 4-2.	.95	32.7/1.5	54
4-4	Paint slightly charred, smooth. No primer showing.	.97	25/2.0	52
4-5	Paint blistered, primer showing 15% of specimen area.	.90	30/2.0	64
4-6	Paint blistered, primer showing 15% of specimen area. Skin delamination in 30% of specimen area.	.90	32.5/2.0	67
4-7	Paint completely removed from 10% of specimen area. Skin delaminated in 50% of specimen area.	.90	30/2.5	82
4-8	Paint completely removed from 75% of specimen area. Skin delaminated in 40% of specimen area.	.75	30/3.0	96
4-9	Paint completely removed from 90% of specimen area. Skin delamination in 50% of specimen area.	.80	30/3.5	109
4-10	Paint completely removed from 90% of specimen area. Skin delamination in 50% of specimen area.	.80	30/4.0	124
4-11	Paint completely removed from 90% of specimen area. Skin delaminated in 60% of specimen area.	.70	30/4.5	136
4-12	Lost entire face sheet--in pieces. A small area first disintegrated; then another small area; then the remaining area delaminated and was drawn into the wind tunnel.	--	30/5.0	153

Table A2-5

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
5-1	Paint lost gloss over most of specimen area--still smooth.	.97	25/1.0	31
5-2	Paint blistered, primer showing in 5% of specimen area.	.95	32.7/1.5	54
5-3	Same as 5-2.	.95	32.5/2.0	67
5-4	Same as 5-2--perhaps slightly more blistering of paint.	.95	30/2.5	82
5-5	Small area of bare aluminum (1% of specimen area). Skin delamination in 50% of specimen area.	.90	30/3.0	96
5-6	Paint completely removed from 40% of specimen area. Skin delamination in 60% of specimen area.	.85	30/3.5	109
5-7	Paint completely removed from 80% of specimen area. Skin delamination in 60% of specimen area.	.85	30/4.0	124
5-8	Paint completely removed from 85% of specimen area. Skin delamination in 70% of specimen area.	.75	30/4.5	136
5-9	Paint completely removed from 90% of specimen area. Skin delamination in 70% of specimen area.	.70	30/5.0	153
5-10	Paint completely removed from 90% of specimen area. Skin delaminated in 60% of specimen area. Piece 1.5 x .7 inch disintegrated/tore out.	.80	28.8/6.0	175

Specimens 5-11 thru 5-12 not exposed.

Table A2-6

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
6-1	Protective paper left on. Specimen discarded.	--	20/1.0	25
6-2	Paint lost gloss over most of specimen--still smooth.	.97	20/1.0	25
6-3	Paint blistered--no primer showing.	.97	25/1.5	41
6-4	Paint blistered. Primer showing in 10% of specimen area.	.95	30/1.5	49
6-5	Paint completely removed from 40% of area. Some degradation (cracking) of skin.	.80	30/2.0	64
6-6	Paint completely removed from 30% of specimen area. Skin delaminated in 50% of specimen area (but no cracks).	.80	30/2.0	64
6-7	Paint completely removed from 95% of specimen area. Skin disintegrated/tore out from 25% of specimen area.	.70	30/2.5	82

Specimens 6-8 to 6-12 not exposed.

Table A2-7

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
7-1	Paint lost gloss over most of specimen area--still smooth.	.97	25/1.0	31
7-2	Paint blistered, primer showing in 5% of specimen area.	.95	30/1.5	49
7-3	Paint completely removed from 20% of specimen area. Slight cracking of aluminum.	.80	30/2.0	64
7-4	Paint completely removed from 75% of specimen area. Small area of skin disintegrated/tore out.	.80	30/2.5	82

Specimens 7-5 to 7-12 not exposed.

Table A2-8

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
8-1	Paint lost gloss over most of specimen area--still smooth.	.97	25/1.0	31
8-2	Paint slightly blistered, primer showing in 2% of specimen area.	.97	30/1.5	49
8-3	Paint blistered, primer showing in 10% of specimen area.	.90	30/2.0	64
8-4	Paint completely removed from 10% of specimen area.	.85	32.5/2.0	67
8-5	Paint completely removed from 10% of specimen area. Skin delamination in 60% of specimen area.	.90	30/2.5	82
8-6	Paint completely removed from 80% of specimen area. Skin disintegrated/tore out in 25% of specimen area.	.70	30/3.0	96

Specimens 8-7 to 8-12 not exposed.

Table A2-9

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
9-1	Paint lost gloss over most of specimen area--still smooth.	.97	25/1.0	31
9-2	Paint blistered, primer showing in 5% of specimen area.	.97	32.7/1.5	54
9-3	Paint completely removed from 10% of specimen area. Primer showing in 5% of specimen area.	.95	32.5/2.0	67
9-4	Paint completely removed from 10% of specimen area. No skin delamination.	.90	30/2.5	82
9-5	Paint completely removed from 20% of specimen area. Skin delamination in 60% of specimen area.	.85	30/3.0	96
9-6	Paint completely removed from 80% of specimen area. Skin disintegrated/tore out, 15% of specimen area.	.85	30/3.5	109

Specimens 9-7 to 9-12 not exposed.

Table A2-10

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
10-1	Paint lost gloss over most of specimen area--still smooth.	.97	25/1.0	31
10-2	Paint slightly blistered. Few small spots of primer showing.	.97	32.7/1.5	54
10-3	Paint blistered, primer showing in 5% of specimen area.	.95	32.5/2.0	67
10-4	Paint blistered, primer showing in 5% of specimen area.	.95	30/2.5	82
10-5	Paint completely removed from 5% of specimen area. Skin delamination in 10% of specimen area.	.95	30/3.0	96
10-6	Paint completely removed from 20% of specimen area. Skin delamination in 70% of specimen area.	.85	30/3.5	109
10-7	Paint completely removed from 40% of specimen area. Skin delamination (in center of specimen) in 60% of specimen area.	.85	30/4.0	124
10-8	Paint completely removed from 50% of specimen area. Skin delamination in 80% of specimen area. 1 x 1 inch area of skin disintegrated/tore out.	.70	30/4.5	136

Specimens 10-9 to 10-12 not exposed.

Table A2-11

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
11-1	Specimen exposed to multiple low level thermal pulses--discarded.	--	5/1.0 10/1.0 15/1.0	6.3 12.8 19
11-2	Paint had very slight loss of gloss toward center of specimen.	.97	15/1.0	19
11-3	Paint lost gloss over most of specimen. Considerable pin-point blistering of paint.	.97	20/1.0	25
11-4	Paint completely removed from 25% of specimen area. FRP not charred.	.85	32.8/1.0	42
11-5	FRP surface charred over 20% of specimen area.	.95	30/2.0	64
11-6	Epoxy burned out of (at least) 1st ply in 60% of specimen area.	.97	32.5/2.0	67
11-7	1st ply disintegrate in 60% of specimen area--some glass fibers attached toward one end. Exposed 2nd ply charred.	.95	30/2.5	82

Specimens 11-8 to 11-12 not exposed.

Table A2-12

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
12-1	Protective paper not removed--specimen discarded.	--	20/1.0	25
12-2	Paint lost gloss over most of specimen area. Considerable pin-point blistering of paint.	.97	20/1.0	25
12-3	Paint blistered over 80% of specimen area.	.97	30/1.0	38
12-4	Paint completely gone, FRP charred toward center of specimen.	.97	30/2.0	64
12-5	Closely similar to 12-4.	.97	32.5/2.0	67
12-6	Protective paper not removed--specimen discarded.	--	30/2.5	82
12-7	Surface of FRP charred/black over entire specimen face.	.97	30/2.5	82
12-8	Surface of FRP charred/black over 95% of specimen area. Delamination (at least 1st ply) in 10% of specimen area.	.95	30/3.0	96
12-9	Surface of FRY charred, 100% of specimen area--carbon beginning to burn off glass (yellowish color) delamination in 10% of area.	.70	30/3.5	109
12-10	1st ply disintegrated in 5% of specimen area--some "tails". Glass cleaner (lighter color) than 12-9. Delamination in 10% of area.	.60	30/4.0	124

Specimens 12-11 to 12-12 not exposed.

Table A-13

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
13-1	Paint lost gloss over most of specimen area. Considerable pin point blistering.	.97	20/1.0	25
13-2	Paint blistered over most of specimen area. Primer showing in 10% of specimen area.	.95	32.8/1.0	42
13-3	FRP exposed, charred/black toward center of specimen.	.97	30/2.0	64
13-4	FRP charred/black, 95% of specimen area. No visable delamination.	.97	32.5/2.0	67
13-5	FRP charred/black, 100% of specimen area. No visable delamination.	.97	30/2.5	82
13-6	FRP charred/black, 95% of specimen area. No visable delamination.	.95	30/3.0	96
13-7	FRP charred (but "cleaner" than 13-6). No visable delamination.	.90	30/3.5	109
13-8	FRP charred ("cleaner" than 13-7) 100% of specimen area. Delamination, 30% of specimen area.	.85	30/4.0	124

Specimens 13-9 to 13-12 cut to include unpainted area.

Table A2-14

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
14-1	Paint lost gloss over most of specimen area. Considerable pin point blistering.	.97	20/1.0	25
14-2	FRP exposed, but not charred, 25% of specimen area.	.85	32.8/1.0	42
14-3	Paint completely removed, FRP charred, 100% of specimen area. Delamination, 40% of area.	.90	30/2.0	64
14-4	Epoxy gone from 1st ply, 40% of specimen area.	.90	32.5/2.0	67
14-5	1st ply disintegrated, 60% of specimen area. Some "tails".	.90	30/2.5	82

Specimens 14-6 to 14-12 not exposed.

Table A2-15

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
15-1	Paint lost gloss over most of specimen area. Considerable pin point blistering.	.97	20/1.0	25
15-2	Paint blistered, 90% of specimen area. Primer showing 1 to 2% of specimen area.	.95	32.8/1.0	42
15-3	Paint completely gone, FRY charred/black. Delamination 20% of specimen area. Some "tails".	.97	30/2.0	64
15-4	Epoxy gone from 1st ply, 40% of specimen area. Glass somewhat "cleaner" than 15-3.	.80	32.5/2.0	67
15-5	1st ply disintegrated, 60% of specimen area. Exposed 2nd ply charred. Some "tails".	.95	30/2.5	82

Specimens 15-6 to 15-12 not exposed.

Table A2-16

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
16-1	Paint lost gloss over most of specimen area. Considerable pin point blistering.	.97	20/1.0	25
16-5	Paint blistered, 90% of specimen area. Primer showing, 1 to 2% of specimen area.	.97	32.8/1.0	42
16-6	Paint completely gone, FRP charred/black. Delamination, 30% of specimen area.	.97	30/2.0	64
16-7	Comparable to 16-6 except no visable delamination.	.97	32.5/2.0	67
16-8	1st ply disintegrated, 25% of specimen area. Exposed 2nd ply charred/black. Some "tails".	.95	30/2.5	82

Specimens 16-2 to 16-4 no good - cut to include unpainted area.

Specimens 16-9 to 16-12 not exposed.

Table A2-17

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
17-1	Protective paper not removed--specimen discarded.	--	20/1.0	25
17-2	Surface of coating liquified, flowed. Some grey/white coating showing.	.90	20/1.0	25
17-3	Most of coating gone. Remaining coating is light grey to white. FRP exposed (not charred), 10% of area.	.60	25/1.5	41
17-4	Similar to 17-3. FRP exposed (but not charred), 15% of specimen area.	.55	30/2.5	82
17-5	Coating mostly gone, 80% of specimen area. FRP not charred.	.60	30/3.5	109
17-6	FRP charred (grey-black), 40% of specimen area. Coating mostly gone from balance of area.	.60	30/4.5	136
17-7	1st ply disintegrated, 70% of specimen area. Exposed 2nd ply charred, delaminated, 10% of specimen area. Some tails.	.85	30/5.0	153

Specimens 17-8 to 17-12 not exposed.

Table A2-18

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
18-1	Large blister formed, flattened, 50% of specimen area--center.	.97	20/1.0	25
18-2	Primer showing (through very thin, remaining black coating), 95% of specimen area.	.80	25/1.5	41
18-3	Primer showing, 100% of specimen area. FRP starting to scorch in center of specimen.	.80	30/2.0	64
18-4	FRP charred, 90% of specimen area. Delamination, 15% of specimen area.	.90	30/2.5	82
18-5	1st ply disintegrated, 45% of specimen area. Exposed 2nd ply charred, some delamination. Some "tails".	.90	30/3.0	96

Specimens 18-6 to 18-12 not exposed.

Table A2-19

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
19-1	No effect - specimen discarded			25 31 38 41
19-2	Large blister (2 inch diameter) formed, flattened to form ridge on downstream edge. No liquification of coating surface.	.30	32.5/2.0	67
19-3	Blistered with coating near FRP-- blistered coating "blew" off. Bare FRP, 10% of specimen area. FRP not charred.	.30	30/2.5	82
19-4	Coating blistered (within white coating near FRP), "blew" off, 50% of specimen area. FRP not charred.	.50	30/3.5	109
19-5	Similar to 19-4 - Coating "blew" off, 75% of specimen area.	.65	30/4.5	136
19-6	Lost most of coating (blistered, "blew" off). FRP charred, 30% of specimen area. Some delamination in charred area.	.75	28.8/6.0	175
19-7	Specimen discarded (coating ink stained).	--	--	--
19-8	1st ply disintegrated, 15% of specimen area. Exposed 2nd ply charred, some delamination. Some white coating still left around upstream edges of specimen.	.90	28.6/7.0	201

Specimens 19-9 to 19-12 not exposed.

Table A2-20

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
20-1	Paint lost gloss over 95% of specimen area. Grey spot in center of specimen, i.e. black partially gone.	.90	25/1.0	31
20-2	Entire specimen area is dark gray, i.e. black anti-static coating is mixing into white base coat.	.85	25/1.5	41
20-3	Entire specimen area is dark grey.	.80	30/2.0	64
20-4	FRP exposed, charred, 50% of specimen area. Rest is dark grey.	.85	30/2.5	82
20-5	FRP exposed, charred 70% of specimen area. No visable delamination.	.85	30/3.0	96
20-6	1st ply disintegrated, 30% of specimen area. Exposed 2nd ply charred, some delamination. Some "tails".	.80	30/3.5	109

Specimens 20-7 to 20-12 not exposed.

Table A2-21

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
21-1	No effect - specimen discarded.	.25	32.5/2.0 30/2.5 30/3.0	67 82 96
21-2	No effect - specimen discarded.	.25	30/3.5 30/4.0 30/4.5	109 124 136
21-3	No effect - specimen discarded.	.25	30/5.0 28.8/6.0 28.6/7.0	153 175 201

Specimens 21-4 to 21-12 not exposed.

Table A2-22

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
22-1	Paint lost gloss over most of specimen area. Still smooth.	.97	20/1.0	25
22-2	Most of topcoat blistered or gone. Primer showing, 90% of specimen area.	.85	25/1.5	41
22-3	Paint mostly gone, FRP charred in center of specimen. No visible delamination.	.95	30/2.0	64
22-4	FRP charred/black, 95% of specimen area. Delamination, 10% of specimen area.	.97	32.5/2.0	67
22-5	1st Ply disintegrated, 50% of specimen area. Exposed 2nd ply charred, no delamination.	.90	30/2.5	82

Specimens 22-6 through 22-12 - Not Exposed

Table A2-23

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
23-1	Paint lost gloss over most of specimen area. Paint slightly roughened.	.97	20/1.0	25
23-2	Paint completely removed, 50% of specimen area.	.90	25/1.5	41
23-3	Most of paint removed. FRP charred. Delamination, 5% of specimen area.	.95	30/2.0	64
23-4	FRP charred, 95% of specimen area. Delamination,10% of specimen area.	.97	32.5/2.0	67
23-5	1st ply disintegrated, 60% of specimen area. Exposed 2nd ply charred. 2nd ply delaminated, 20% of specimen area. Some "tails".	.95	30/2.5	82

Specimens 23-6 through 23-12 - Not Exposed

Table A2-24

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
24-1	Paint lost gloss over most of specimen area. Slightly roughened	.97	20/1.0	25
24-2	Paint blistered. Primer showing in 20% of specimen area.	.90	32.7/1.5	54
24-3	FRP charred/black toward center of specimen area. No delamination.	.95	30/2.0	64
24-4	FRP charred/black, 100% of specimen area. No visible delamination.	.95	30/2.5	82
24-5	FRP charred/black, 95% of specimen area. No visible delamination.	.97	32.5/2.0	67
24-6	FRP charred/black over most of specimen area. Center of specimen is becoming yellowish, i.e. carbon is burning off glass.	.85	30/3.0	96
24-7	FRP charred, 100% of specimen area. Delamination, 20% of specimen area.	.85	30/3.5	109
24-8	Delaminated, 50% of specimen area. Otherwise, same as 24-7.	.85	30/4.0	124
24-9	Delaminated, 80% of specimen area. Glass fabric relatively clean (and still intact) toward center.	.65	30/4.5	136
24-10	Delamination, 60% of specimen area. Very similar to 24-9.	.70	30/5.0	153

Specimens 24-11 and 24-12 - Not Available for Thermal Exposure

Table A2-25

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
25-1	Paint Lost Gloss Over Most of Specimen Area. Some Fine Blistering Toward Center of Specimen	.97	30/1.0	38
25-2	Paint Blistered. Primer showing in 10% of Specimen Area.	.95	30/2.0	64
25-3	Paint Blistered. Primer Showing in 10% of Specimen Area.	.85	30/2.5	82
25-4	Paint Completely Removed in 30% of Specimen Area. Skin Delamination, 60% of Specimen Area.	.85	30/4.0	124

Specimens 25-5 through 25-12 - Not Exposed

Table A2-26

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
26-1	Paint Lost Gloss, Blistered in 50% of Specimen Area.	.97	30/1.0	38
26-2	Similar to 26-1. Primer showing in 5% of specimen area.	.95	30/2.0	64
26-3	Almost identical to 26-2.	.90	30/2.5	82
26-4	Primer showing (small spots) in 20% of specimen area. Skin delamination, 50% of specimen area.	.90	30/4.0	124
26-5	No visible skin delamination. Otherwise similar to 26-4.	.90	30/4.5	136
26-6	Lost entire face sheet toward end of run. Face sheet is warped but in one piece.	.85	30/5.0	153

Specimens 26-7 through 26-12 - Not Exposed

Table A2-27

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
27-1	Paint lost gloss, blistered over 50% of specimen area.	.97	30/1.0	38
27-2	Similar to 27-1. Primer showing (small spots) in 2% of area.	.95	30/2.0	64
27-3	Paint blistered over most of specimen. Some whitish spots (but no primer) showing.	.85	30/2.5	82
27-4	Primer showing (small spots) in 5% of specimen area. No bare aluminum and no delamination.	.95	30/4.0	124
27-5	Primer showing (small spots) in 25% of specimen area. Very small area of skin delamination.	.90	30/4.5	136
27-6	Paint completely removed, 30% of area. Skin delamination 30% of specimen area-in center of specimen face.	.85	30/5.0	153
27-7	Paint completely removed, 60% of specimen area. Skin delamination in 70% of specimen area.	.80	28.8/6.0	175
27-8	Paint completely removed, 30% of specimen area. Skin delamination, 90% of specimen area.	.85	28.6/7.0	201

Specimens 27-9 through 27-12 - Not Exposed

Table A2-28

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
28-1	90% of paint "blew" off. Metal is clean and bright.	.25 (metal) .97 (paint)	25/1.0	31
28-2	100% of paint "blew" off. Metal is clean and bright.	.25	30/2.0	64
28-3	Entire face sheet delaminated at skin/adhesive interface. Face has very little warpage.	.25	30/2.5	82

Specimens 28-4 through 28-12 - Not Exposed

Table A2-29

SPECIMEN EXPOSURE RECORD

SPECIMEN NUMBER	EFFECTS OF THERMAL EXPOSURE	FINAL ABSORPTIVITY (ESTIMATED)	PEAK FLUX/ PULSE DURATION (SECONDS)	INCIDENT FLUENCE (cal/cm^2)
29-1	Paint blistered - primer showing in a few small spots.	.95	25/1.0	31
29-2	Paint completely removed, 40% of specimen area. Primer showing (small spots) in 10% of specimen area.	.95	30/1.5	49
29-3	Paint completely removed, 100% of specimen area. Ply 1 delaminated, 10% of specimen area.	.90	32.5/2.0	67
29-4	Ply 1 delaminated, 50% of specimen area. Some yellowish color toward center of specimen.	.70	30/2.5	82
29-5	Same as 29-4.	.70	30/3.0	96
29-6	Ply 1 delaminated, 60% of specimen area. Less yellowish color.	.80	30/3.5	109
29-7	Ply 1 delaminated, 40% of specimen area.	.90	30/4.0	124
29-8	Ply 1 delaminated, 80% of specimen area.	.85	30/4.5	136
29-9	Very similar to 29-8.	.90	30/5.0	153
29-10	Very similar to 29-8, except for some "fuzzing" of Ply 1 graphite fibers at downstream edge of specimen.	.85	28.8/6.0	175

Specimens 29-11 and 29-12 - Not Available for Exposure

APPENDIX 3

WIND TUNNEL DRAG TESTS

This appendix contains the results of each run made on the thermally damaged specimens. Twenty-three groups of specimens were tested at Mach numbers of .5 and .8 and in addition two groups were run at M = 1.2. Drag measurements were made for each run and reduced to a skin friction coefficient and equivalent sand roughness. Between 5 and 15 runs were made on each sample and the results were averaged over these runs. Where the data scatter was greater than about $\pm$ 30% no average was taken. Generally this excessive data scatter could be traced to an unusual surface condition, such as debonding or a large ridge of paint etc.

Table A3-1: TABULATED DATA, 1 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 12 MIL ALUMINUM SKIN
COATING: 3 MIL BLACK ENAMEL (MIL-C-83286), 250°F CURE TEMP.

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
1-2	24	132	0.5	4.07	3.557	0.82	
		133	0.5	7.54	3.031	0.33	C_f: Average = 3.284 x 10^{-3}
		134	0.8	5.66	3.584	0.86	
		135	0.8	10.68	3.119	0.40	K_{seq} Average = 0.559 x 10^{-4} inches
		172	0.5	4.13	3.470	0.75	
		185	0.8	10.59	3.188	0.45	
		186	0.8	5.86	3.327	0.61	
		187	0.8	9.10	3.248	0.50	
		188	0.5	4.14	3.334	0.60	
		189	0.5	7.60	3.203	0.50	
		203	0.5	4.21	3.189	0.47	
		209	0.5	4.26	3.216	0.50	
		210	0.5	7.73	3.220	0.50	
		211	0.8	5.89	3.351	0.63	
		212	0.8	10.61	3.217	0.47	
1-3	41	136	0.5	4.19	2.755	Smooth	
		137	0.5	7.47	2.544	Smooth	
		138	0.8	6.01	3.200	0.47	C_{f_i} Average = 3.437 x 10^{-3}
		139	0.8	10.77	3.074	0.38	
		190	0.5	4.17	3.641	1.00	K_{seq} Average = 0.875 x 10^{-3} inches
		191	0.5	4.15	3.677	1.04	
		192	0.5	7.63	3.653	1.03	
		193	0.8	5.83	3.956	1.38	
		195	0.8	5.82	3.753	1.09	
		197	0.5	4.14	3.519	0.85	
		198	0.5	7.69	3.417	0.73	
		199	0.8	5.85	3.204	0.49	
		200	0.8	10.62	3.787	0.99	
		201	0.8	5.83	3.667	0.98	
		202	0.8	10.64	3.706	0.94	

Table A3-2: TABULATED DATA, 3 SERIES

CONSTRUCTION: SAME AS 1-X SERIES EXCEPT ALUMINUM SKIN THICKNESS IS 20 MILS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
3-1	30	260	0.5	3.978	3.713	1.20	
		262	0.5	4.088	3.476	0.78	C_{f_i} Avg. = 3.665×10^{-3}
		263	0.5	7.105	3.688	1.05	
		265	0.8	5.733	3.680	0.99	K_{seq} Avg. = 0.999×10^{-3} inches
		266	0.8	10.290	3.949	1.22	
		267	0.5	4.051	3.506	0.80	
		268	0.5	7.203	3.644	0.95	
3-2	47	269	0.5	4.07	4.011	1.56	C_{f_i} Avg. = 4.29×10^{-3}
		270	0.5	7.14	4.224	1.70	
		271	0.8	5.70	4.471	2.25	K_{seq} Avg. = 1.92×10^{-3} inches
		272	0.8	10.22	4.522	2.17	

Table A3-3: TABULATED DATA, 4 SERIES

CONSTRUCTION: SAME AS 1-X EXCEPT ALUMINUM SKIN THICKNESS IS 25 MILS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
4-1		SPECIMEN DESTROYED - NO DATA					
4-2	52.5	276	0.5	4.01	3.768	1.20	C_{f_i} Avg. = 3.76×10^{-3} not incl run 277
		277	0.5	7.12	2.858	0.10	
		278	0.8	5.70	3.749	1.07	K_{seq} Avg. = 1.083×10^{-3} inches not incl run 277
		279	0.8	10.24	3.764	0.98	
4-3	52	280	0.5	4.075	4.019	1.56	C_{f_i} Avg. = 3.665×10^{-3}
		281	0.5	7.106	4.280	1.83	
		283	0.8	5.711	4.468	2.25	K_{seq} Avg. = 0.999×10^{-3} inches
		284	0.8	10.270	4.440	1.91	
4-5		SPECIMEN DESTROYED - NO DATA					

Table A3-4: TABULATED DATA, 5 SERIES

CONSTRUCTION: SAME AS 1-X SERIES EXCEPT ALUMINUM SKIN THICKNESS IS 32 MILS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
5-1	30	288	0.5	3.95	2.462	Smooth	Suspicious Drag Traces
		289	0.5	7.06	2.415	Smooth	
		291	0.8	5.61	2.194	Smooth	
		292	0.8	10.13	2.564	Smooth	
		294	0.8	5.57	3.669	1.00	
5-2	52	296	0.5	3.89	3.724	1.13	C_{f_i} Avg. = 3.815×10^{-3} not incl 300 or 303
		297	0.5	6.98	3.811	1.13	
		298	0.5	6.98	4.044	1.43	
		300	0.8	5.60	4.221	1.76	K_{seq} Avg. = 1.148×10^{-3} inches not incl 300 or 303
		301	0.8	10.21	3.796	1.00	
		302	0.8	5.63	3.699	1.05	
		303	0.5	7.05	3.216	0.52	
5-3	64	304	0.5	3.91	3.367	0.68	C_f Avg. = 3.927×10^{-3} not incl run 304
		305	0.5	7.05	3.891	1.20	
		306	0.8	5.65	3.841	1.20	K_{seq} Avg. = 1.243×10^{-3} inches not incl run 304
		307	0.8	10.17	4.050	1.33	
5-4	78	308	0.5	3.88	3.799	1.22	C_f Avg. = 3.792×10^{-3} not incl 311
		309	0.5	7.04	3.715	1.05	
		310	0.8	5.62	3.800	1.14	K_{seq} Avg. = 1.133×10^{-3} inches not incl 311
		311	0.8	10.20	3.436	0.67	
		312	0.8	8.73	3.855	1.12	

Table A3-5: TABULATED DATA, 10 SERIES

CONSTRUCTION: SAME AS 5-X SERIES EXCEPT 350^{o}F CURE TEMP.

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
10-1	30	313	0.5	4.08	3.328	0.60	
		314	0.5	6.05	3.621	0.94	Large Data Scatter
		315	0.5	7.05	4.200	1.68	
		316	0.8	5.66	4.400	2.15	Shows Mach Number Dependence
		317	0.8	8.79	4.492	2.13	
		318	0.8	10.21	4.522	2.18	
10-2	52	319	0.5	3.92	3.724	1.12	
		320	0.5	6.00	3.798	1.14	C_f Avg. = 3.94×10^{-3}
		321	0.5	7.06	3.920	1.25	
		322	0.8	5.67	3.986	1.42	K_s Avg. = 1.33×10^{-3} Inches
		323	0.8	8.76	4.296	1.72	
10-3	64	325	0.5	4.01	3.660	1.05	
		326	0.5	6.12	3.816	1.15	C_f Avg. = 3.78×10^{-3} not incl. 329 or 331
		327	0.5	7.11	3.841	1.15	
		328	0.8	7.76	4.232	1.62	K_s Avg. = 1.24×10^{-3} inches not incl. 329 or 331
		329	0.8	8.88	4.825	2.87	
		331	0.8	10.34	5.028	3.35	
10-4	79	333	0.5	4.02	3.847	1.28	
		334	0.5	6.11	3.815	1.20	C_f Avg. = 3.74×10^{-3}
		335	0.5	7.09	3.839	1.15	
		336	0.8	5.81	3.751	1.07	K_s Avg. = 1.06×10^{-3} inches
		337	0.8	8.87	3.519	0.75	
		338	0.8	10.35	3.688	0.90	

Table A3-6: TABULATED DATA, 11 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 13.5 MIL EPOXY-FIBERGLAS SKIN, 350^{o}F CURE TEMP.
COATING: ENAMEL (MIL-C-83286), BLACK, 3 MIL NOMINAL THICKNESS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
11-3	24	339	0.5	4.08	3.720	1.10	
		340	0.5	6.17	3.842	1.18	C_{f_i} Avg. = 3.77×10^{-3}
		341	0.5	7.19	3.757	1.08	
		342	0.8	5.78	3.722	1.01	
		343	0.8	8.87	3.897	1.14	K_{seq} Avg. = 1.07×10^{-3} inches
		344	0.8	10.35	3.668	0.88	
11-4	39	345	0.5	4.03	4.244	1.97	
		346	0.5	6.09	4.396	2.08	C_{f_i} Avg. = 4.38×10^{-3}
		347	0.5	7.18	4.338	1.90	
		348	0.8	5.78	4.358	2.05	K_{seq} Avg. = 2.02×10^{-3} inches
		349	0.8	8.89	4.451	2.07	
		350	0.8	10.37	4.470	2.05	

Table A3-7: TABULATED DATA, 12 SERIES

CONSTRUCTION: SAME AS 11-X EXCEPT EPOXY FIBERGLAS SKIN IS 20 MILS THICK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	$10^3\ C_{f_i}$	K_{seq} (10^3 Inches)	Comments
12-2	24	351	0.5	4.05	3.406	0.70	
		352	0.5	6.15	3.287	0.55	C_{f_i} Avg. = 3.20×10^{-3}
		353	0.5	7.13	3.290	0.55	
		354	0.8	5.74	3.377	0.63	
		355	0.8	8.85	2.944	0.30	K_{seq} Avg. = 0.50×10^{-3} inches
		357	0.8	10.38	2.919	0.27	
12-3	37	358	0.5	4.05	4.451	2.40	
		359	0.5	6.19	4.575	2.45	C_{f_i} Avg. = 4.96×10^{-3}
		360	0.5	7.19	4.919	3.15	
		361	0.8	5.77	5.286	4.40	K_{seq} Avg. = 3.42×10^{-3} inches based on average C_{f_i} value
		362	0.8	8.93	5.295	4.08	
		363	0.8	10.44	5.222	3.85	
12-4	62	364	0.5	4.07	4.160	1.84	
		365	0.5	6.22	4.358	2.07	C_{f_i} Avg. = 4.526×10^{-3}
		366	0.5	7.34	4.230	1.73	
		367	0.8	5.83	4.695	2.73	
		368	0.8	8.98	4.853	2.95	K_{seq} = 2.35×10^{-3} inches based on average C_{f_i} value
		369	0.8	10.51	4.862	2.92	
12-7	80	371	0.5	4.08	4.117	1.75	
		372	0.5	6.26	4.189	1.70	C_{f_i} Avg. = 4.293×10^{-3}
		373	0.5	7.25	4.182	1.60	
		374	0.8	5.84	4.390	2.10	K_{seq} Avg. = 1.867×10^{-3} inches
		377	0.8	8.72	4.321	1.80	
		378	0.8	10.23	4.556	2.24	

Table A3-8: TABULATED DATA, 13 SERIES

CONSTRUCTION: SAME AS 11-X EXCEPT EPOXY FIBERGLAS SKIN IS 30 MILS THICK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	$10^3 C_{f_i}$	K_{seq} (10^3 Inches)	Comments
13-1	24	379	0.5	3.90	3.919	1.92	
		380	0.5	6.06	3.957	1.89	
		381	0.5	7.06	4.042	1.92	C_{f_i} Avg. = 4.17×10^{-3}
		382	0.8	5.61	4.300	1.93	
		383	0.8	8.72	4.394	1.90	K_{seq} Avg. = 1.92×10^{-3} inches
		384	0.8	9.88	4.407	1.95	
13-3	61	385	0.5	3.88	4.125	1.79	
		386	0.5	6.03	4.280	1.88	
		388	0.5	7.04	4.731	2.70	C_{f_i} Avg. = 4.89×10^{-3} not incl. 385, 386
		389	0.8	5.54	4.788	2.90	
		390	0.8	8.63	4.942	3.15	K_{seq} Avg. = 3.05×10^{-3} inches not incl. 385, 386
		391	0.8	10.14	5.100	3.45	
13-7	102	392	0.5	3.90	4.823	3.27	
		394	0.5	6.52	3.928	1.83	C_{f_i} Avg. = 4.23×10^{-3}
		395	0.5	7.62	3.431	0.73	
		396	0.8	5.58	4.526	2.36	
		397	0.8	8.67	4.322	1.78	K_{seq} = 1.78×10^{-3} inches based on average C_{f_i} value
		398	0.8	10.27	4.337	1.80	

Table A3-9 : TABULATED DATA, 14 SERIES

CONSTRUCTION: SAME AS 11-X EXCEPT CURE TEMPERATURE IS 250^{o}F

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
14-1	24	403	0.5	7.114	4.314	1.90	
		404	0.5	3.98	4.428	2.35	C_{f_i} Avg. = 4.719 x 10^{-3}
		405	0.5	6.06	4.612	2.50	
		406	0.8	5.58	4.793	3.00	K_{seq} = 2.78 x 10^{-3} inches based on average C_{f_i} value
		407	0.8	8.74	5.077	3.50	
		408	0.8	10.22	5.089	3.43	
14-2	38	409	0.5	3.96	4.324	2.13	
		410	0.5	6.04	4.524	2.34	C_{f_i} Avg. = 4.40 x 10^{-3} not incl. run 413
		411	0.5	7.89	4.381	2.25	
		412	0.8	5.58	4.377	2.08	K_{seq} Avg. = 2.20 x 10^{-3} inches not incl. run 413
		413	0.8	8.68	3.893	1.15	

Table A3-10: TABULATED DATA, 15 SERIES

CONSTRUCTION: SAME AS 14-X EXCEPT EPOXY-FIBERGLAS SKIN IS 20 MILS THICK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
15-1	24	415	0.5	3.94	4.550	2.60	
		416	0.5	6.10	4.728	2.77	C_{f_i} Avg. = 4.993×10^{-3}
		417	0.5	7.12	4.870	3.03	
		418	0.8	5.70	5.165	4.05	
		420	0.8	10.34	5.312	4.20	K_{seq} = 3.47×10^{-3} inches based on average C_{f_i} value
		421	0.8	8.87	5.335	4.15	
15-2	40	422	0.5	4.00	4.435	2.36	
		423	0.5	6.16	4.490	2.26	C_{f_i} Avg. = 4.746×10^{-3}
		424	0.5	7.23	4.512	2.23	
		425	0.8	5.83	4.750	2.88	
		426	0.8	8.92	4.978	3.23	K_{seq} = 2.83×10^{-3} inches based on average C_{f_i} value
		427	0.8	10.38	5.308	4.05	
15-5	79	SAMPLE SKIN FABRIC LOST - NO DATA					

Table A3-11: TABULATED DATA, 15 SERIES

CONSTRUCTION: SAME AS 14-X EXCEPT EPOXY-FIBERGLAS SKIN IS 30 MILS THICK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
16-1	24	428	0.5	3.99	4.390	2.28	
		429	0.5	6.12	4.458	2.23	
		430	0.5	7.15	4.546	2.58	C_{f_i} Avg. = 4.604 x 10^{-3}
		431	0.8	5.77	4.781	2.85	
		432	0.8	8.89	5.070	3.45	
		433	0.8	10.37	5.162	3.47	K_{seq} = 2.50 x 10^{-3} inches based on average C_{f_i} value
		760	1.22	6.27	4.260	1.49	
		761	1.22	9.59	4.354	1.05	
		762	1.22	12.78	4.454	0.70	
16-5	41	441	0.5	4.00	3.924	1.42	
		442	0.5	6.08	4.203	1.75	C_{f_i} Avg. = 4.39 x 10^{-3} not incl run 765
		443	0.5	7.15	4.261	1.77	
		444	0.8	5.78	4.216	1.76	
		445	0.8	8.87	4.483	2.10	K_{seq} Avg. = 1.85 x 10^{-3} inches not incl run 765
		446	0.8	10.39	4.469	2.07	
		763	1.22	6.27	4.763	2.30	
		764	1.22	9.59	4.829	1.62	
		765	1.22	12.75	4.512	0.75	
16-6	62	434	0.5	4.05	5.344	4.92	
		435	0.5	6.06	5.519	4.95	C_{f_i} Avg. = 5.58 x 10^{-3}
		436	0.5	7.19	5.361	4.42	
		437	0.8	5.73	5.601	5.53	
		438	0.8	8.89	5.845	5.93	K_{seq} = 5.31 x 10^{-3} inches based on average C_{f_i} value
		440	0.8	10.39	5.817	5.78	

Table A3-12: TABULATED DATA, 17 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 30 MIL EPOXY-FIBERGLAS SKIN, 250°F CURE TEMP.
COATING: ASTROCOAT, BLACK OVER WHITE, 15 MIL NOMINAL THICKNESS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
17-2	23	447	0.5	4.03	3.601	0.95	
		448	0.5	6.22	3.596	0.85	C_{f_i} Avg. = 3.92 x 10^{-3} not incl runs 450 or 705
		449	0.5	7.22	3.713	1.00	
		450	0.8	5.80	4.239	1.83	
		701	1.22	6.38	4.214	1.43	K_{seq} Avg. = 1.09 x 10^{-3} inches not incl runs 450 or 705
		702	1.22	9.70	4.472	1.20	
		705	1.22	12.86	3.867	0.30	
17-3	32	455	0.5	4.30	4.440	2.40	
		456	0.5	6.20	5.088	3.54	Excessive Data Scatter
		457	0.5	7.09	5.536	4.95	No Average Possible
		458	0.8	5.73	5.672	5.80	
		459	0.8	8.87	6.067	6.75	
		745	1.22	6.31	5.738	4.45	
		747	1.22	9.61	5.735	3.22	
		748	1.22	12.82	6.172	3.00	
17-4	62	462	0.5	3.92	5.913	7.27	
		463	0.5	6.02	6.165	7.45	
		464	0.5	7.08	5.198	7.45	Excessive Data Scatter
		465	0.8	5.61	6.621	9.75	
		466	0.8	8.73	6.955		
		467	0.8	10.24	7.515		
		749	1.22	6.27	5.815	4.68	
		750	1.22	9.61	5.856	3.55	
		751	1.22	12.79	6.300	3.27	

Table A3-12: TABULATED DATA, 17 SERIES (CONTINUED)

CONSTRUCTION: HONEYCOMB SANDWICH, 30 MIL EPOXY-FIBERGLAS SKIN, 250^{O}F CURE TEMP.
COATING: ASTROCOAT, BLACK OVER WHITE, 15 MIL NOMINAL THICKNESS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
17-5	86	468	0.5	3.92	3.855	1.33	
		469	0.5	6.07	4.788	2.92	Excessive Data Scatter
		470	0.5	7.09	4.899	3.09	
		471	0.8	5.68	5.043	3.63	
		473	0.5	4.24	3.930	1.45	
		752	1.22	6.27	6.013	5.20	
		754	1.22	9.61	6.231	4.47	
		755	1.22	12.76	6.450	3.60	
17-6	107	481	0.5	3.98	5.079	4.01	
		482	0.5	6.19	5.133	3.93	
		483	0.5	7.11	5.451	4.73	Excessive Data Scatter
		485	0.8	5.65	5.495	5.13	
		486	0.8	8.79	5.716	5.46	
		487	0.8	10.37	6.152	7.12	
		756	1.22	6.27	5.982	5.10	
		758	1.22	9.51	6.290	4.65	
		759	1.22	12.71	5.595	1.95	
17-7	76 to 150	644	0.5	3.99	23.90		
		645	0.5	6.09	20.75		Specimen continually lost loose fiberglas, therefore no constant drag value
		647	0.5	7.09	14.87		
		648	0.5	6.06	15.05		
		649	0.5	4.02	15.96		
		651	0.5	7.13	7.63		

Table A3-13: TABULATED DATA, 18 SERIES

CONSTRUCTION: SAME AS 17-X EXCEPT ASTROCOAT IS ALL BLACK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
18-1	24	632	0.8	5.589	7.233		
		633	0.8	8.711	7.694		Excessive Data Scatter
		634	0.5	3.910	6.799		
		635	0.5	6.187	6.975		
		636	0.5	7.233	6.716		
18-2	36	474	0.5	3.871	4.52		
		475	0.5	6.053	5.05		Excessive Data Scatter
		476	0.5	7.089	5.23		
		479	0.8	5.653	7.48		
18-3	56	507	0.8	8.59	6.021	6.65	
		508	0.8	10.07	6.368	7.93	Excessive Data Scatter
		509	0.8	5.58	5.575	5.39	Debonded over central
		510	0.5	6.74	5.927	6.78	quarter.
		511	0.5	6.02	5.313	4.50	
		512	0.5	3.83	5.226	4.54	
18-4	77	488	0.5	3.93	5.088	4.05	
		489	0.5	6.05	5.230	4.20	
		490	0.5	7.15	5.224	4.00	Excessive Data Scatter
		491	0.8	5.66	7.349		Debonded over central
		494	0.8	8.52	6.174	7.78	quarter.
		495	0.8	10.02	6.315	7.93	
		496	0.8	5.46	6.152	7.74	
		497	0.5	7.36	4.788	2.85	
		498	0.5	5.89	5.472	5.08	
		499	0.5	3.84	5.218	4.20	

Table A3-14: TABULATED DATA, 19 SERIES

CONSTRUCTION: SAME AS 17-X EXCEPT ASTROCOAT IS ALL WHITE

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
19-5	111	501	0.8	8.54	5.599	5.05	
		502	0.8	9.98	5.639	5.20	
		503	0.8	5.40	5.700	5.90	Excessive Data Scatter
		504	0.5	7.02	5.112	3.70	Thick paint flow
		505	0.5	6.01	4.943	3.38	
		506	0.5	3.82	4.919	3.55	
19-6	88 to 170	513	0.8	8.61	5.495	4.65	
		514	0.8	10.07	5.246	3.90	
		515	0.8	5.49	5460	5.03	Debonded over aft quarter
		516	0.5	7.05	5.139	3.70	
		517	0.5	6.03	4.939	3.35	
		518	0.5	3.83	4.933	3.63	

Table A3-15 : TABULATED DATA, 20 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 30 MIL EPOXY FIBERGLASS SKIN, 250^{0} CURE TEMP.
COATING: FLOROCARBON, BLACK OVER WHITE, 15 MIL NOMINAL THICKNESS

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
20-1	28	521	0.8	8.75	4.079	1.40	
		522	0.8	10.18	4.033	1.33	C_{f_i} Avg. = 4.15×10^{-3}
		523	0.8	5.69	4.345	2.02	
		524	0.5	7.15	4.121	1.55	
		525	0.5	5.99	4.228	1.75	K_{seq} Avg. = 1.63×10^{-3} inches
		526	0.5	3.91	4.105	1.75	
20-3	58	527	0.8	8.66	4.150	1.50	
		528	0.8	9.92	4.526	2.20	C_{f_i} Avg. = 4.10×10^{-3}
		529	0.5	7.02	3.695	100	
		530	0.5	5.94	4.032	1.45	
		531	0.5	3.85	3.850	1.33	K_{seq} Avg. = 1.61×10^{-3} inches
		532	0.8	5.55	4.371	2.17	
20-5	88	533	0.8	8.66	5.398	4.36	
		536	0.8	10.16	5.014	3.28	C_{f_i} Avg. = 5.01×10^{-3}
		537	0.8	5.52	5.080	3.76	
		538	0.5	6.67	4.930	3.20	
		539	0.5	6.03	4.970	3.42	K_{seq} Avg. = 3.48×10^{-3} inches
		540	0.5	3.88	4.663	2.88	

Table A3-16: TABULATED DATA, 21 SERIES

CONSTRUCTION: SAME AS 20-X SERIES EXCEPT FLOROCARBON IS ALL WHITE

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
21-2	34	541	0.8	8.58	3.667	0.90	
		542	0.8	10.37	3.898	1.14	C_{f_i} Avg. = 4.19 x 10
		543	0.8	5.75	4.361	2.08	
		544	0.5	7.29	4.159	1.62	
		545	0.5	6.23	4.340	1.95	
		546	0.5	4.29	4.206	1.92	K_{seq} Avg. = 1.73 x 10^{-3} inches
		547	0.8	8.69	4.691	2.52	

Table A3-17: TABULATED DATA, 22 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 30 MIL EPOXY FIBERGLAS SKIN, 250^{0} CURE TEMP.
COATING: 3 MIL (NOMINAL) BLACK LACQUER (MIL-L-81352)

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
22-3	62	548	0.8	8.91	5.144	3.68	
		549	0.8	10.42	5.140	3.65	C_{f_i} Avg. = 4.79×10^{-3}
		550	0.8	5.82	4.852	3.15	
		551	0.5	7.30	4.730	2.70	
		552	0.5	6.25	4.609	2.52	K_{seq} Avg. = 3.04×10^{-3} inches
		553	0.5	4.12	4.271	2.53	

Table A3-18: TABULATED DATA, 23 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 30 MIL EPOXY FIBERGLAS SKIN, 250^{o} CURE TEMP.
COATING: 3 MIL (NOMINAL) BLACK POLYSULFIDE ENAMEL PRIMER

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
23-1	24	555	0.8	8.93	5.58	5.03	
		556	0.8	10.45	5.46	4.65	Excessive Data Scatter
		557	0.8	5.79	4.93	3.34	
		558	0.5	7.28	4.76	3.37	
		559	0.5	6.20	4.67	2.66	
		560	0.5	4.04	4.35	2.15	
23-2	39	561	0.8	8.99	4.455	2.05	
		562	0.8	10.49	4.520	2.20	C_{f_i} Avg. = 4.40 x 10^{-3}
		563	0.8	5.81	4.423	2.20	
		564	0.5	7.39	4.377	1.95	
		565	0.5	6.28	4136	1.62	K_{seq} Avg. = 2.07 x 10^{-3} inches
		566	0.5	4.04	4.461	2.42	
23-3	62	567	0.8	8.96	4.770	2.73	
		568	0.8	10.46	5.071	3.50	C_{f_i} Avg. = 4.41 x 10^{-3} not incl run 568
		570	0.8	10.44	4.689	2.55	
		571	0.8	5.82	4.352	2.05	
		572	0.5	4.32	3.978	1.49	K_{seq} Avg. = 2.12 x 10^{-3} inches not incl run 568
		573	0.5	7.41	4.314	1.85	
		574	0.5	6.29	4.384	2.03	
23-5	78	663	0.8	10.42	8.538		
		665	0.8	5.76	8.044		High drag values - cannot get good equivalent Ks.
		666	0.5	4.06	6.792		
		667	0.5	7.22	7.239		

Table A3-19: TABULATED DATA, 24 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, 50 MIL EPOXY - "S" FIBERGLAS SKIN, 350^{0} CURE TEMP.
COATING: 3 MIL (NOMINAL) BLACK ENAMEL (MIL-C-83286)

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
24-5	65	587	0.8	8.76	4.234	1.63	
		588	0.8	10.21	4.771	2.75	C_f Avg. = 4.68 x 10^{-3} not incl. run 587
		589	0.8	5.63	4.576	2.49	
		590	0.5	7.02	4.772	2.80	
		591	0.5	6.05	4.644	2.58	K_{seq} Avg. = 2.68 x 10^{-3} inches not incl run 587
		592	0.5	3.86	4.627	2.77	
24-9	122	653	0.8	10.42	5.419	4.43	
		654	0.8	5.81	5.065	3.73	C_{f_i} Avg. = 5.00 x 10^{-3}
		655	0.5	7.16	4.767	2.82	
		656	0.5	3.93	4.733	3.02	K_{seq} = 3.43 x 10^{-3} inches based on average C_{f_i} value

Table A3-20: TABULATED DATA, 25 SERIES

CONSTRUCTION: 40 MIL ALUMINUM SHEET
COATING: 3 MIL (NOMINAL) BLACK ENAMEL (MIL-C-83286)

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
25-3	78	575	0.8	8.90	4.020	1.30	
		576	0.8	10.43	3.878	1.13	C_{f_i} Avg. = 3.81×10^{-3}
		577	0.8	5.76	3.768	1.10	
		578	0.5	7.33	3.773	1.08	
		579	0.5	6.23	3.772	1.06	K_{seq} Avg. = 1.11×10^{-3} inches
		580	0.5	4.06	3.637	0.97	

Table A3-21: TABULATED DATA, 26 SERIES

CONSTRUCTION: SAME AS 25-X, EXCEPT ALUMINUM SKIN IS 50 MILS THICK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
26-1	37	581	0.8	8.65	2.944	0.28	
		582	0.8	10.17	3.412	0.65	
		583	0.8	5.61	3.216	0.48	C_{f_i} Avg. = 3.26×10^{-3}
		584	0.5	7.06	3.333	0.62	
		585	0.5	6.01	3.327	0.60	
		586	0.5	3.88	3.304	0.59	K_{seq} Avg. = 0.58×10^{-3} inches
		669	0.5	6.20	3.404	0.68	
		670	0.5	7.22	3.474	0.78	
		671	0.5	4.09	3.335	0.62	
		672	0.8	8.96	3.522	0.75	
		673	0.8	10.42	3.191	0.45	
		674	0.8	5.80	3.202	0.47	
26-3	62	593	0.8	8.68	3.960	1.16	
		594	0.8	10.15	3.535	0.75	C_{f_i} Avg. = 3.72×10^{-3} not incl run 598
		595	0.8	5.60	3.626	0.92	
		596	0.5	7.06	3.664	0.94	
		579	0.5	5.99	3.692	0.99	K_{seq} Avg. = 0.98×10^{-3} inches not incl run 598
		598	0.5	3.92	3.118	0.36	
		599	0.8	8.69	3.868	1.13	
26-6	145	657	0.5	4.06	14.13		The skin was completely removed leaving exposed honeycomb.
		658	0.5	7.21	14.33		
		659	0.8	5.78	15.58		
		661	0.8	10.32	17.25		

Table A3-22: TABULATED DATA, 27 SERIES

CONSTRUCTION: SAME AS 25-X EXCEPT ALUMINUM SKIN IS 63 MILS THICK

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
27-6	138	601	0.8	8.66	5.020	3.38	
		602	0.8	5.58	4.684	2.73	C_{f_i} Avg. = 4.65 x 10^{-3} not incl run 606
		603	0.5	3.90	4.437	2.35	
		604	0.5	6.09	4.498	2.26	
		605	0.5	7.09	4.634	2.43	K_{seq} Avg. = 2.63 x 10^{-3} inches not incl run 606
		606	0.8	10.18	5.336	4.15	

Table A3-23: TABULATED DATA, 29 SERIES

CONSTRUCTION: HONEYCOMB SANDWICH, EPOXY GRAPHITE SKIN, 250^{o}F CURE TEMP.
COATING: 3 MIL (NOMINAL) BLACK ENAMEL (MIL-C-83286)

Specimen	Absorbed Fluence (cal/cm^2)	Run No.	Nominal Mach No.	Re/L (10^{-6} Ft^{-1})	10^3 C_{f_i}	K_{seq} (10^3 Inches)	Comments
29-1	30	609	0.8	8.73	5.571	4.96	
		610	0.8	5.61	5.063	3.74	
		611	0.5	3.99	5.327	4.85	
		612	0.5	6.10	5.832	6.33	
		613	0.5	7.14	5.970	6.60	
		614	0.8	10.13	5.382	4.28	
29-2	47	615	0.8	8.64	5.442	4.47	
		616	0.8	5.59	5.319	4.55	C_{f_i} Avg. = 5.38 x 10^{-3}
		617	0.5	3.96	5.163	4.35	
		618	0.5	6.09	5.398	4.75	
		619	0.5	7.11	5.423	4.67	K_{seq} = 4.60 x 10^{-3} inches
		620	0.8	10.20	5.531	4.80	

APPENDIX 4

SURFACE PROFILE TRACES

The traces in this appendix were made on a "Tallysurf" profile measuring machine. This machine traces out the profile to a variety of magnifications. Those used here ranged (vertically) from 10,000 (for the smooth plates) to 500 (for blistered paint). The horizontal magnification used was generally 20 times, so that 2" on the profile corresponds to 0.1" on the sample.

The "centerline average" was also measured and calculated for most of these samples by the profilometer. This is a measure of the local roughness over a small traverse - 0.15". Heights are measured at 5 points on this traverse (.030" apart) and the "CLA" is the average absolute distance from the mean of these points. It does not appear to be a useful measure of roughness for this current investigation.

For most samples the profiles were measured along the centerline about 1/4 and 3/4 the distance from the leading edge. The following specimens were measured:

Smooth Surfaces (Unexposed)		CLA	Max. Roughness
10-1	Enamel on Al	2 micro ins.	5 micro ins.
27	Polished Al Test Sample	6 micro ins.	40 micro ins.
27-11	Unpolished Aluminum	35 micro ins.	250 micro ins.
11-8	Enamel on Fiberglass	35 micro ins.	300 micro ins.
21-4	Fluorcarbon on Fiberglass	35 micro ins.	250 micro ins.

Exposed Aluminum/Enamel		CLA	Max. Roughness
4-4	25 mil Skin	180 μ ins.	1.5×10^{-3} ins.
4-4	25 mil Skin	200 μ ins.	2.5×10^{-3} ins.
10-1	32 mil Skin	100 μ ins.	0.5×10^{-3} ins.
10-2	32 mil Skin	140 μ ins.	1.2×10^{-3} ins.
10-3	32 mil Skin	150 μ ins.	1.2×10^{-3} ins.
10-4	32 mil Skin	200 μ ins.	1.0×10^{-3} ins.

Exposed Fiberglass/Enamel		CLA	Max. Roughness
12-2	20 mil Skin	80 μ ins.	$.8 \times 10^{-3}$ ins.
12-3	20 mil Skin	175 μ ins.	1.5×10^{-3} ins.
12-4	20 mil Skin	300 μ ins.	2.0×10^{-3} ins.
12-7	20 mil Skin	260 μ ins.	2.0×10^{-3} ins.
11-3	13.5 mil Skin. Pin-paint blistering	--	3×10^{-3} ins.
14-1	13.5 mil Skin. Pin-paint blistering	--	$4\text{-}5 \times 10^{-3}$ ins.
29-1	20 mil Graphite Skin	200 μ ins.	4×10^{-3} ins.
29-2	20 mil Graphite Skin	290 μ ins.	2×10^{-3} ins.

The final page in this appendix is the report describing the measurements made in the Particle Identification Lab of the sandpapers used to "calibrate" the drag of the samples.

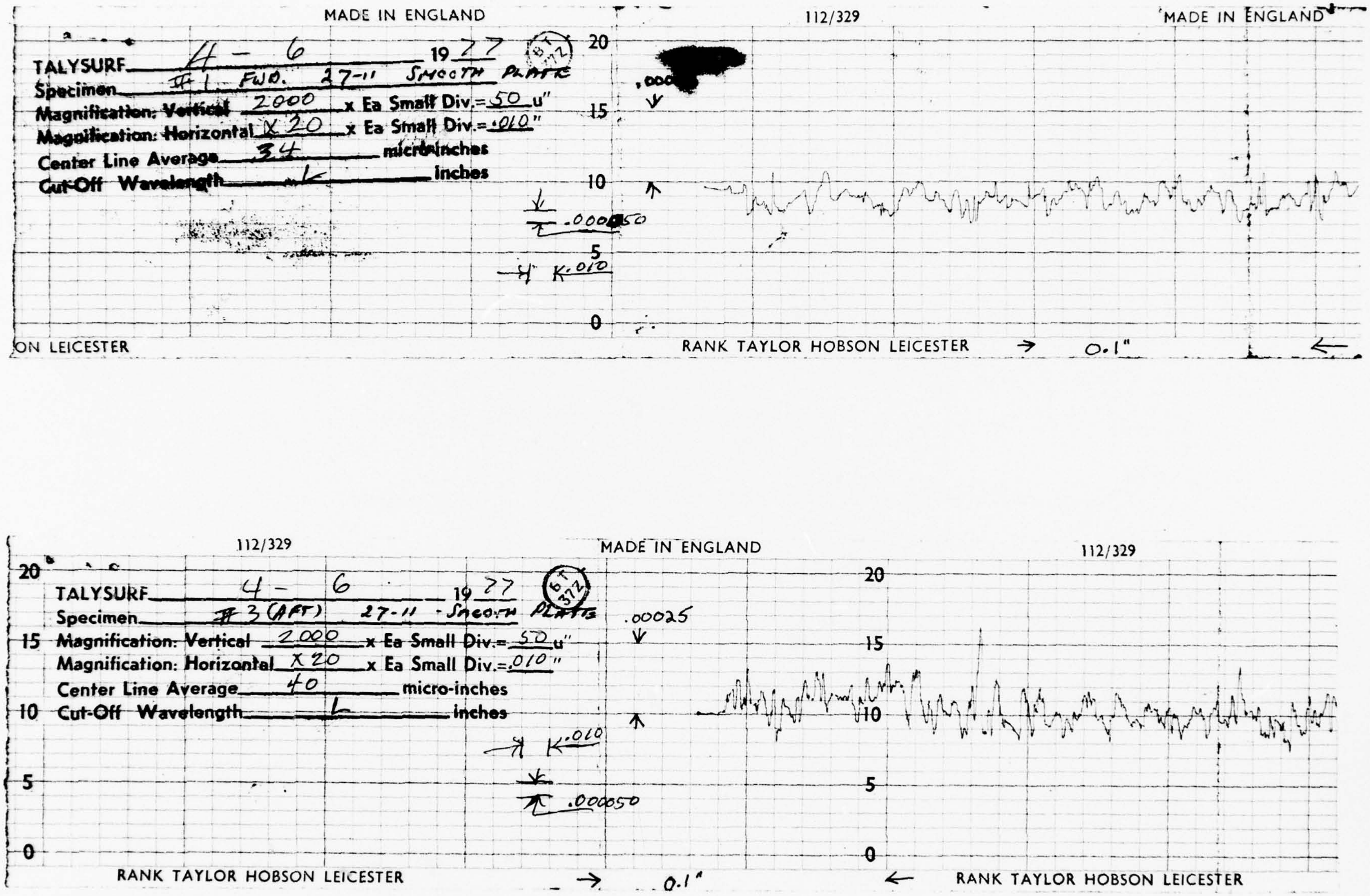


Fig. A4-1 Specimen 27-11. Smooth Polished Aluminum Test Plate (63 mils thick)

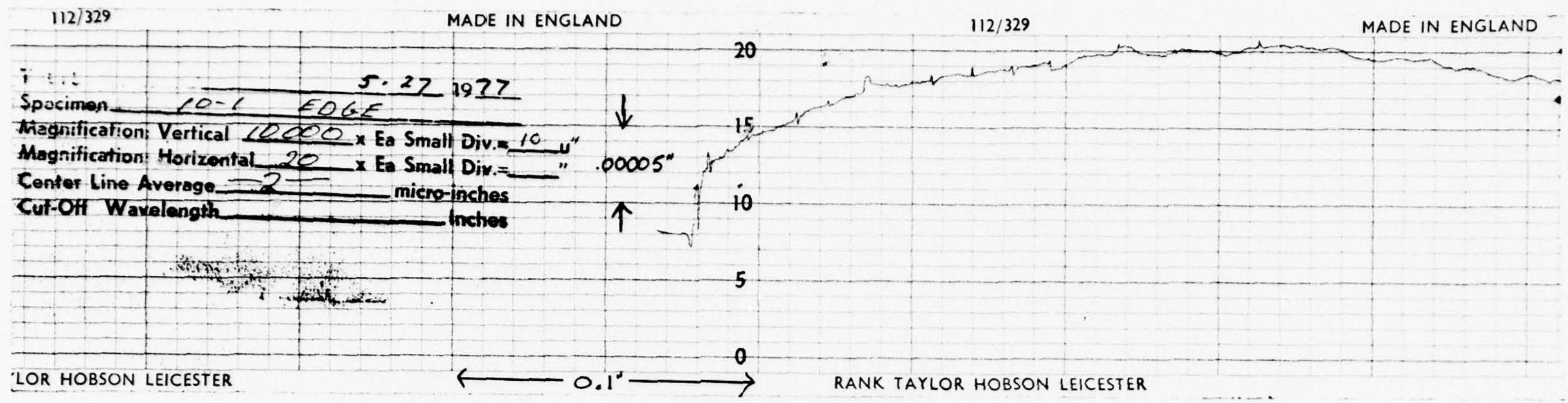


Fig. A4-2 (a) Specimen 10-1. Profile of Smooth Enamel Over 32 mil Aluminum Skin

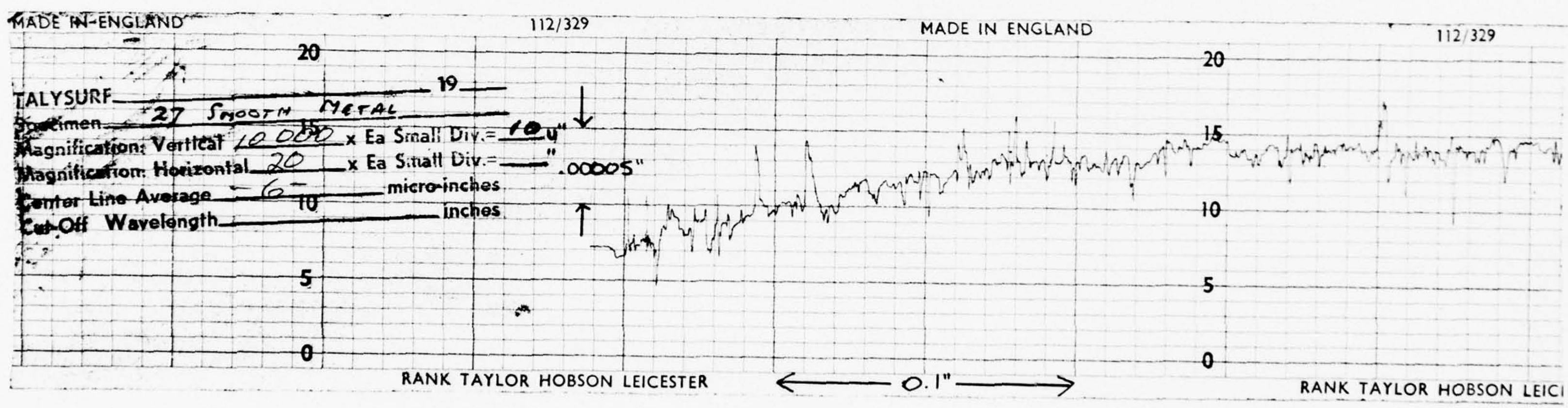


Fig. A4-2 (b) Specimen 27. Smooth Unpolished Aluminum Plate (63 mils)

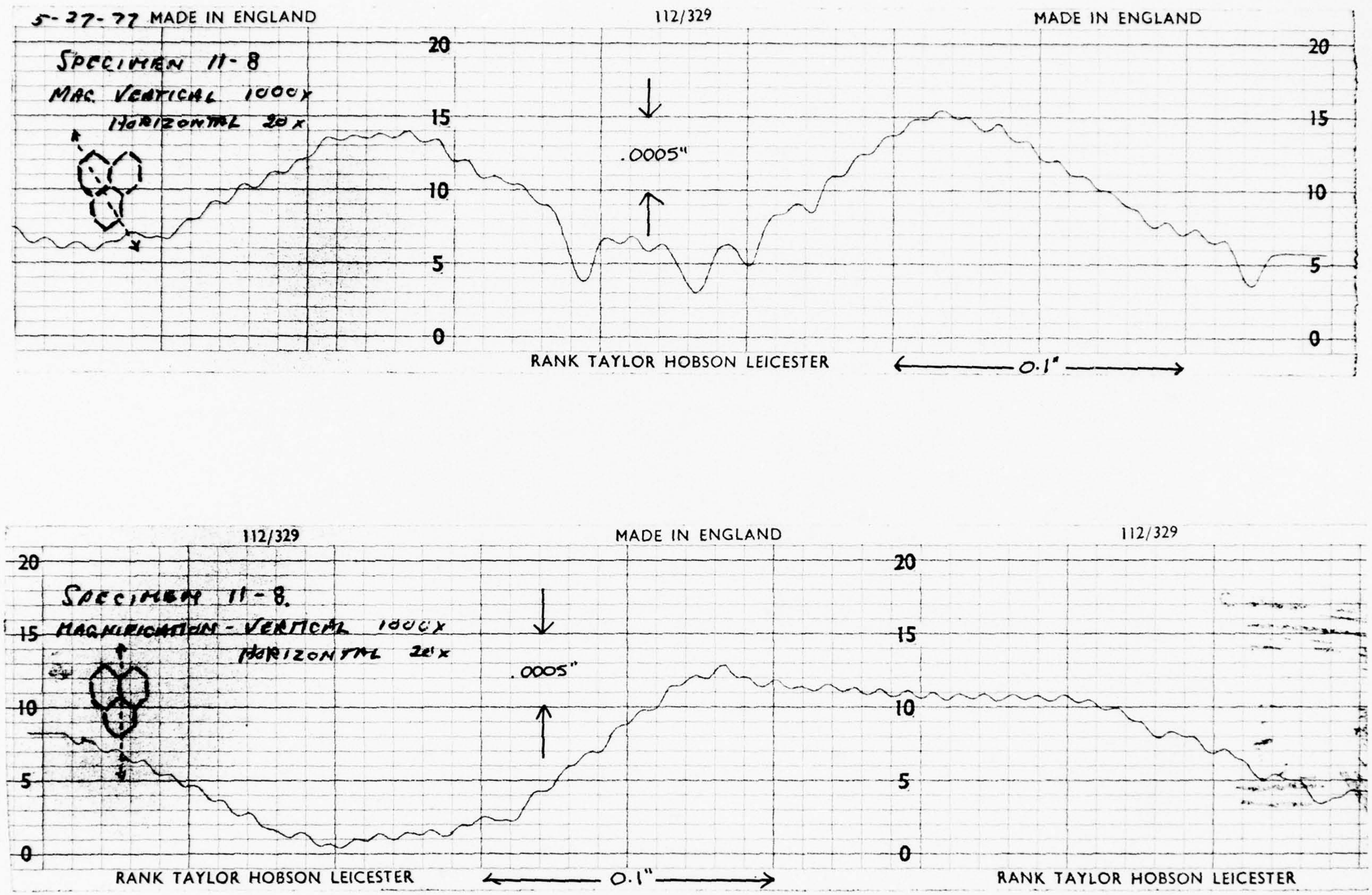


Fig. A4-3 Specimen 11-8. Unexposed Enamel over 13.5 mil Fiberglass Skin. Profile Traverses in directions indicated with respect to honeycomb core.

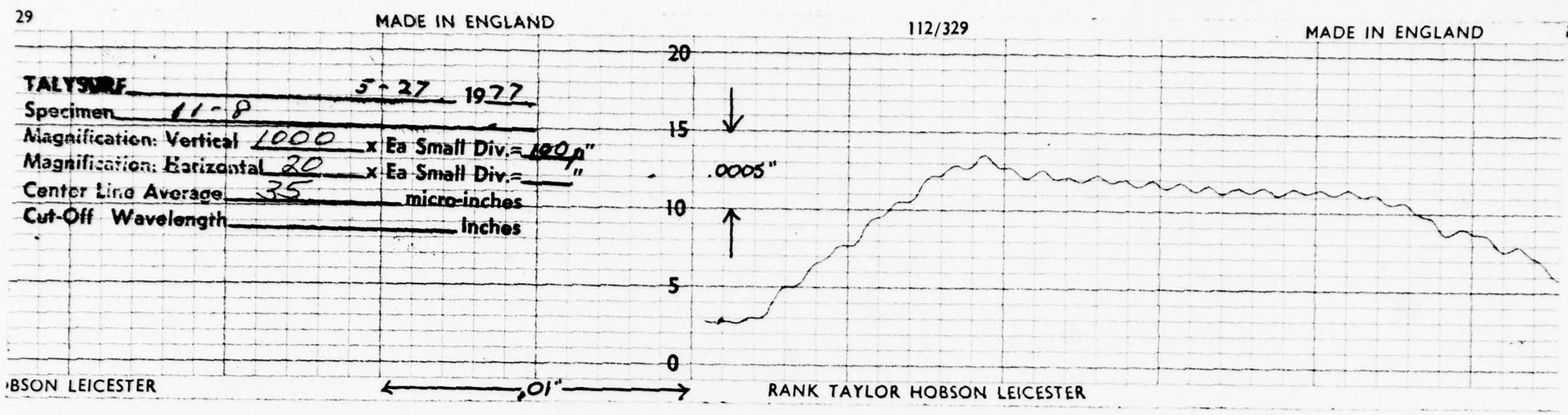


Fig. A4-4 (a) Specimen 11-8. (As in A4-3) Traverse was in an arbitrary direction.

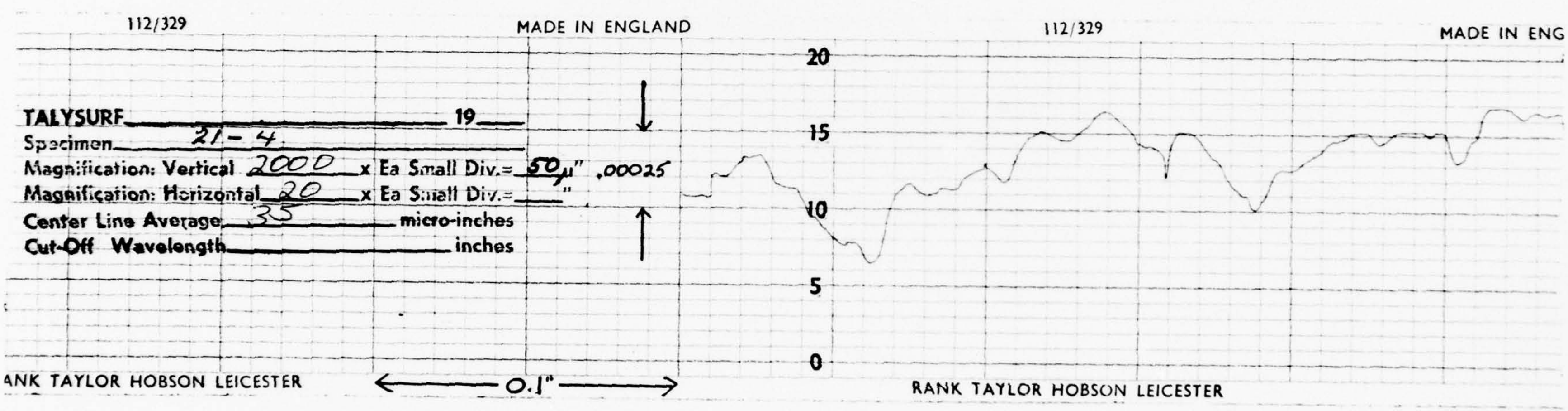
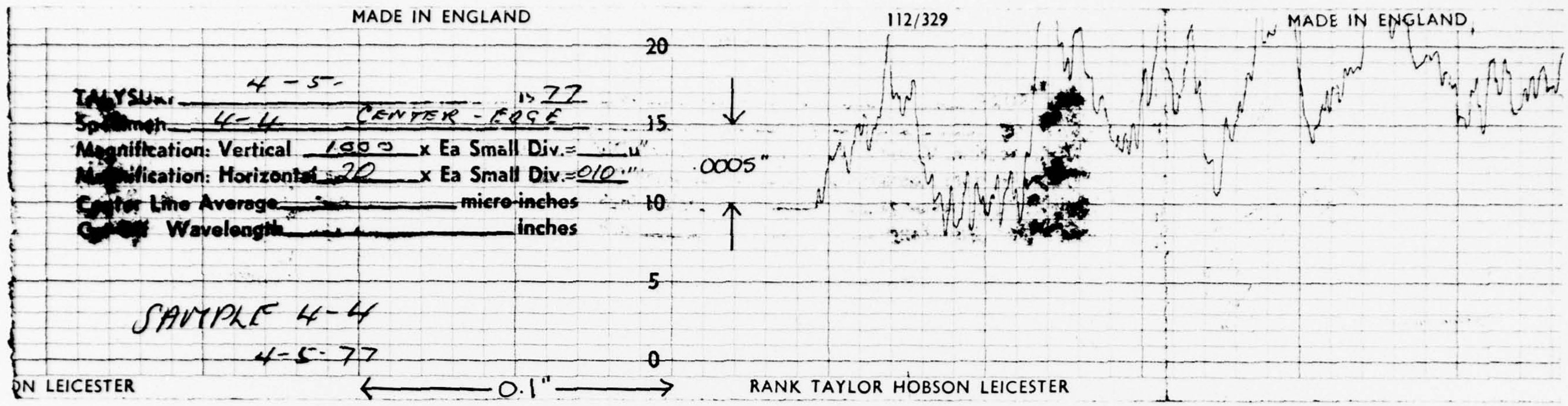


Fig. A4-4 (b) Specimen 21-4. Unexposed fluorcarbon (15 mil) over 30 mil fiberglass skin. Honeycomb is identical to 4 (a).



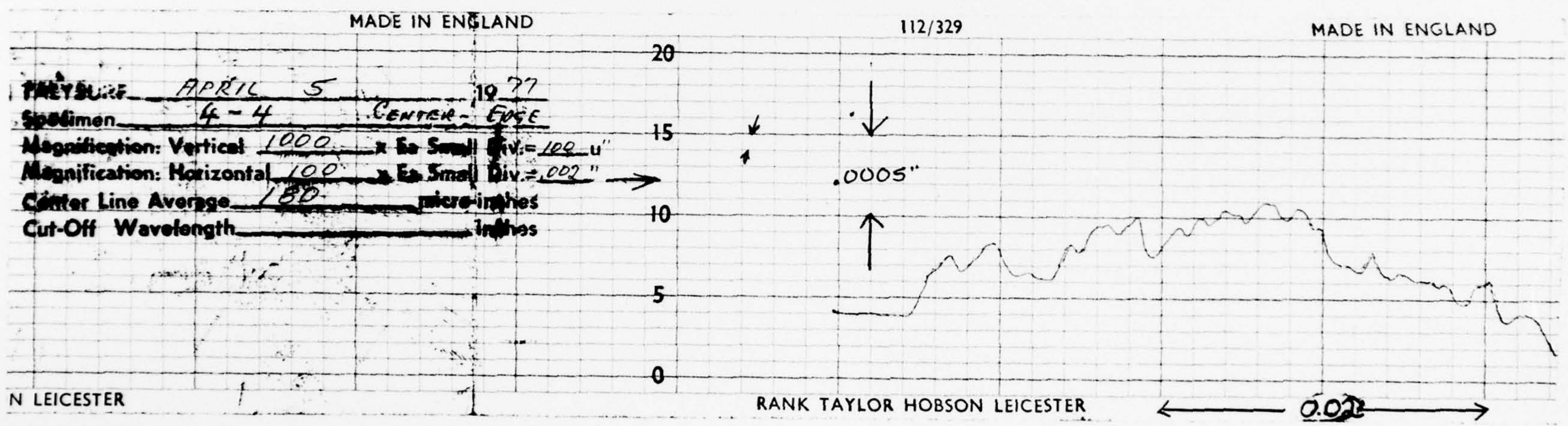


Fig. A4-5. Specimen 4-4 at edge. Exposed enamel (52 cals/cm^2) over 25 mil aluminum skin. Note difference in horizontal scale.

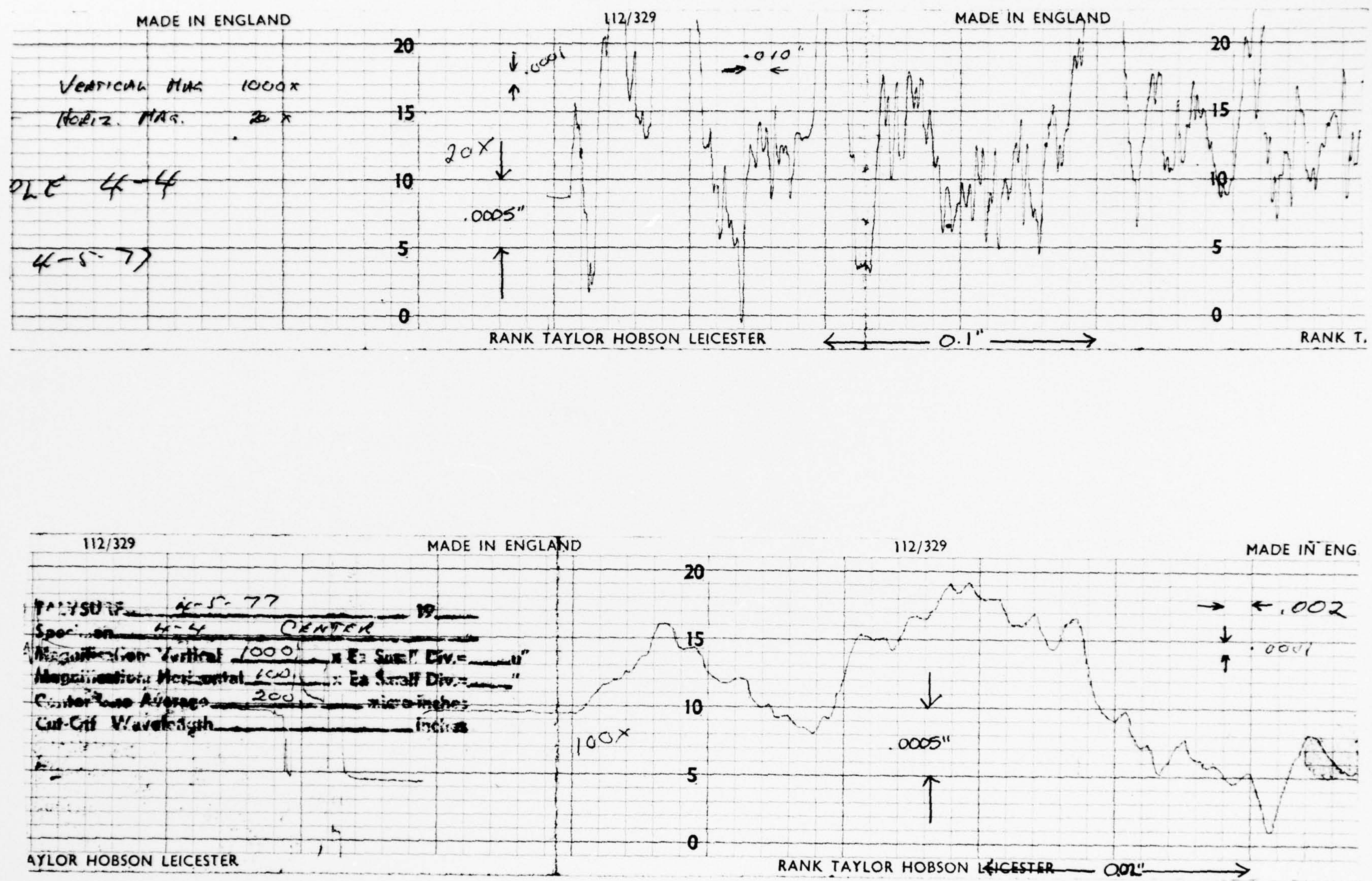
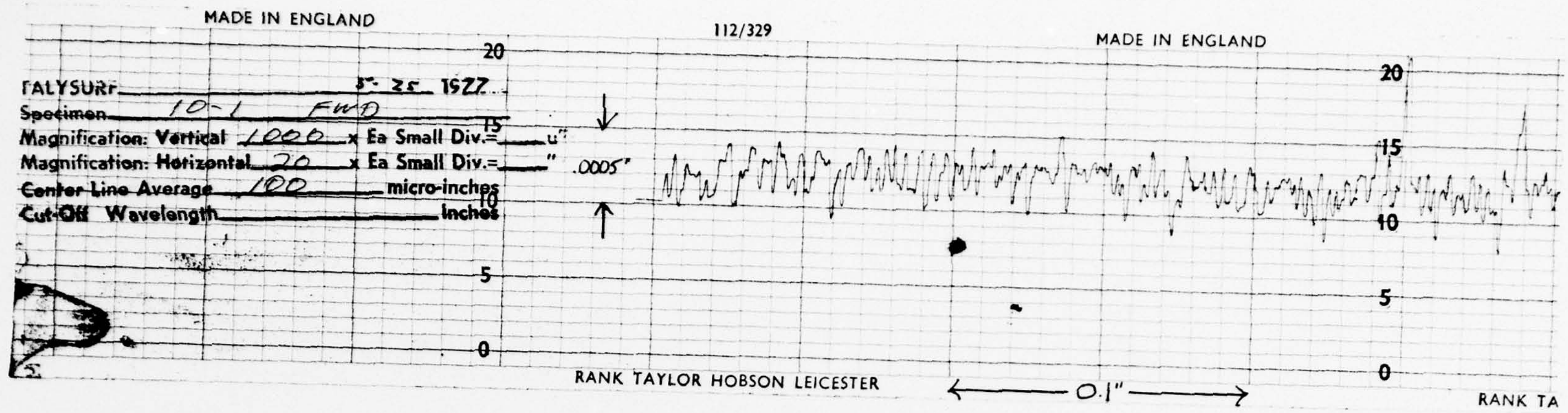


Fig. A4-6 Same as -5, but profile taken in center.



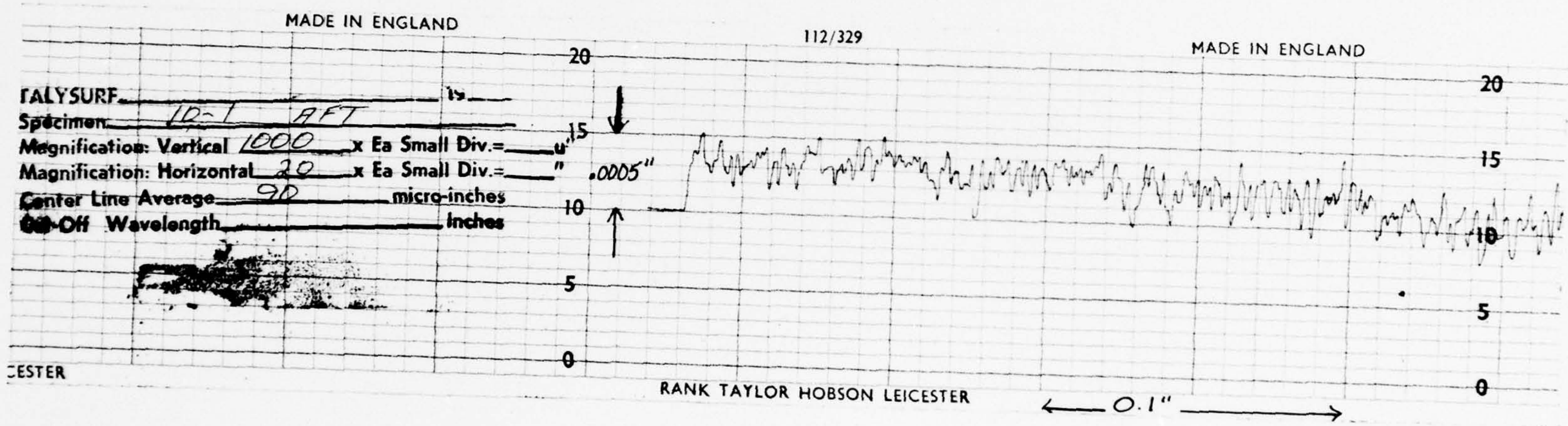
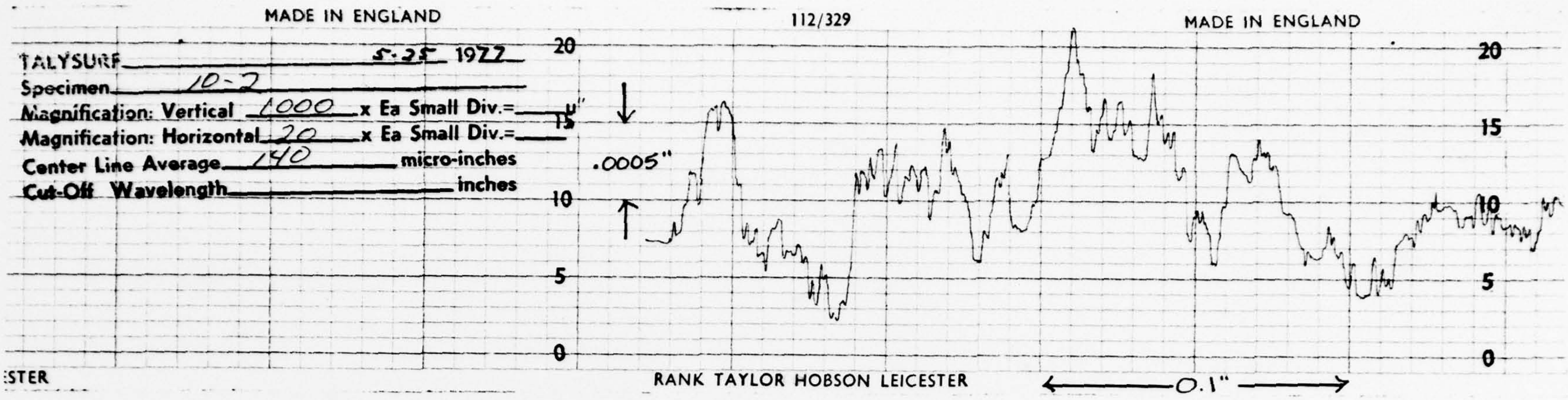


Fig A4-7. Specimen 10-1. Exposed enamel (31 cals/cm^2) over 32 mil aluminum skin.



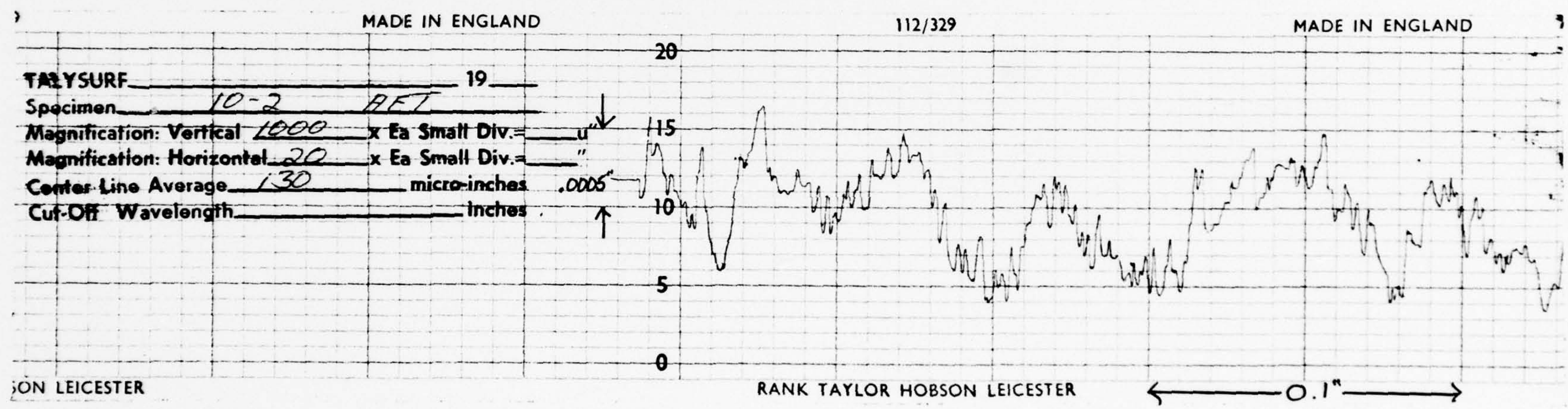
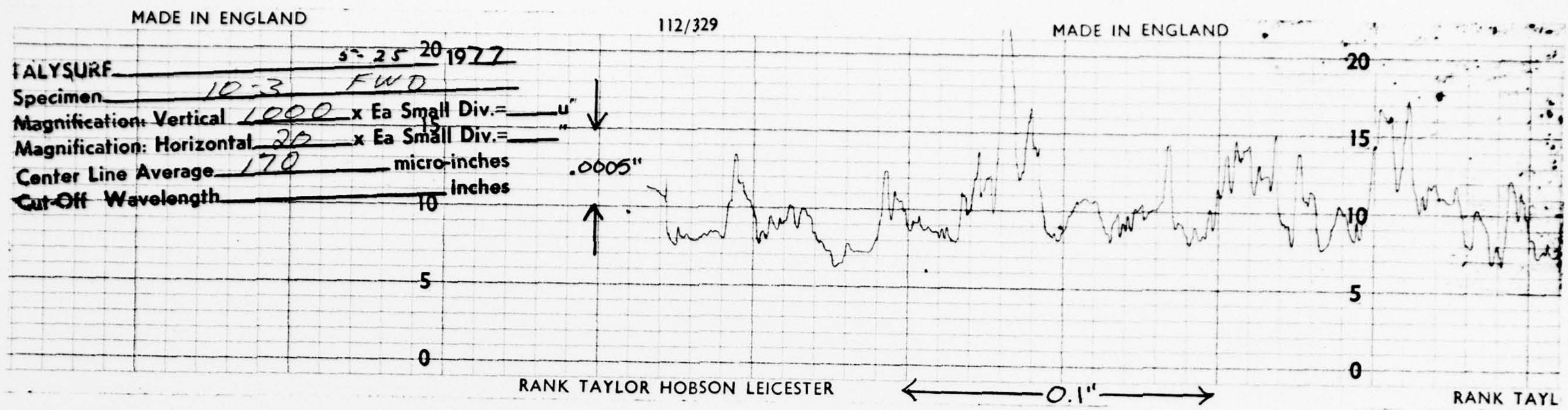


Fig. A4-8. Specimen 10-2. Fluence 54 cals/cm^2.



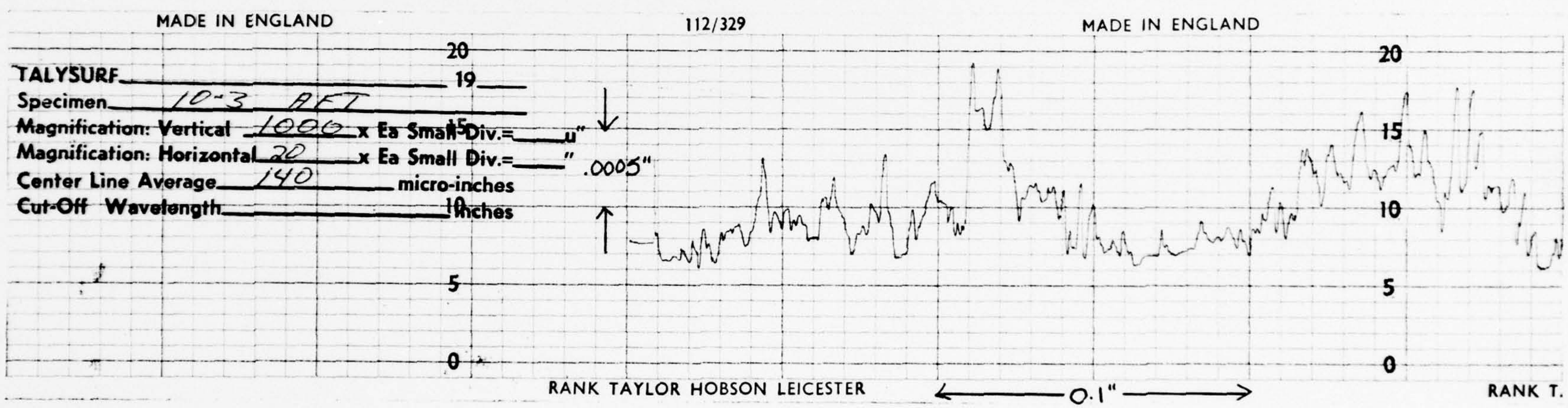


Fig. A4-9. Specimen 10-3. Fluence 67 cals/cm^2.

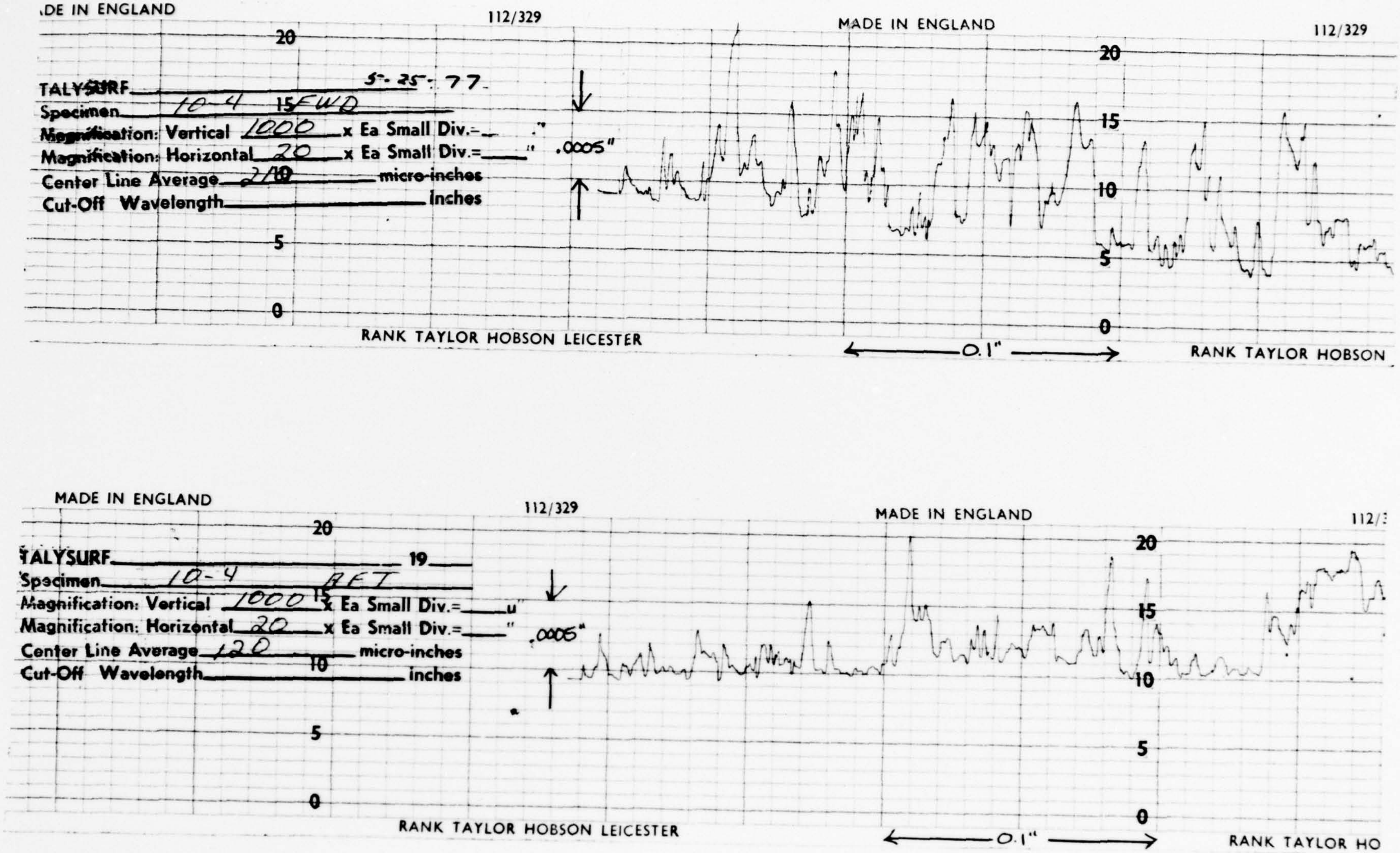
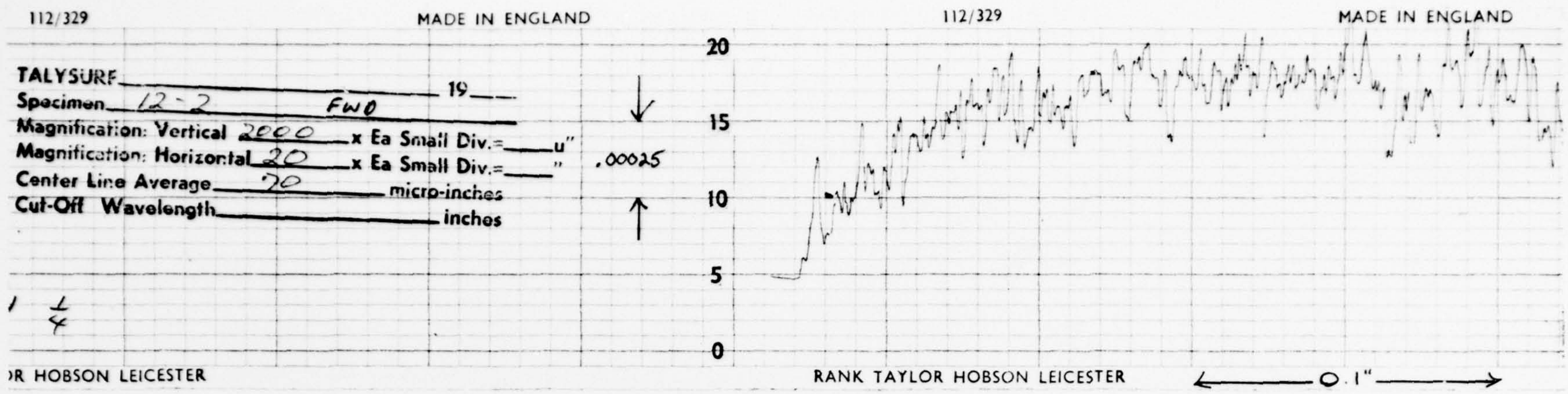


Fig. A4-10 Specimen 10-4. Fluence 82 cals/cm^2.



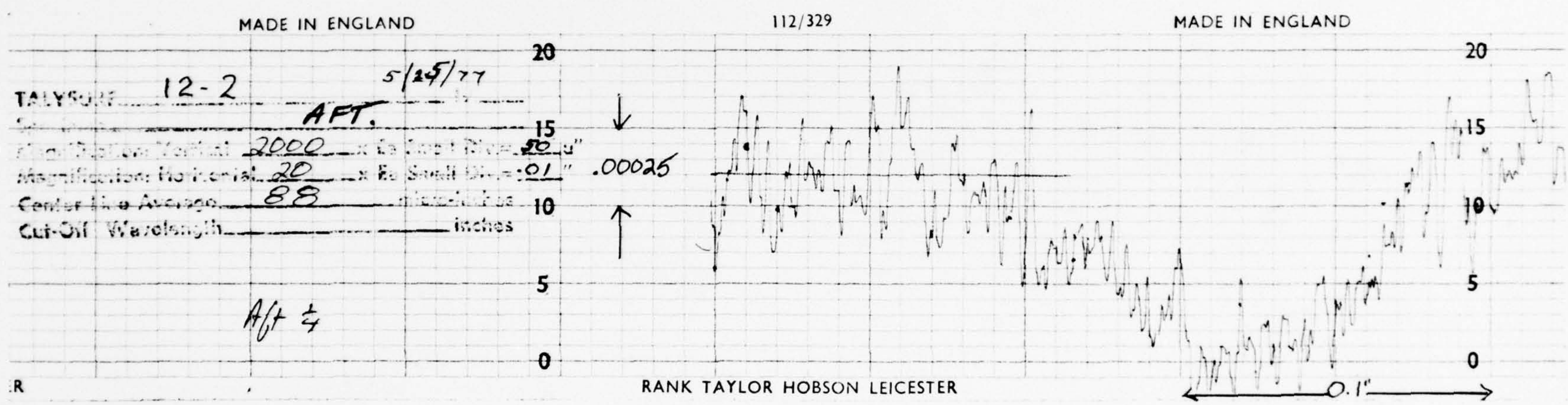
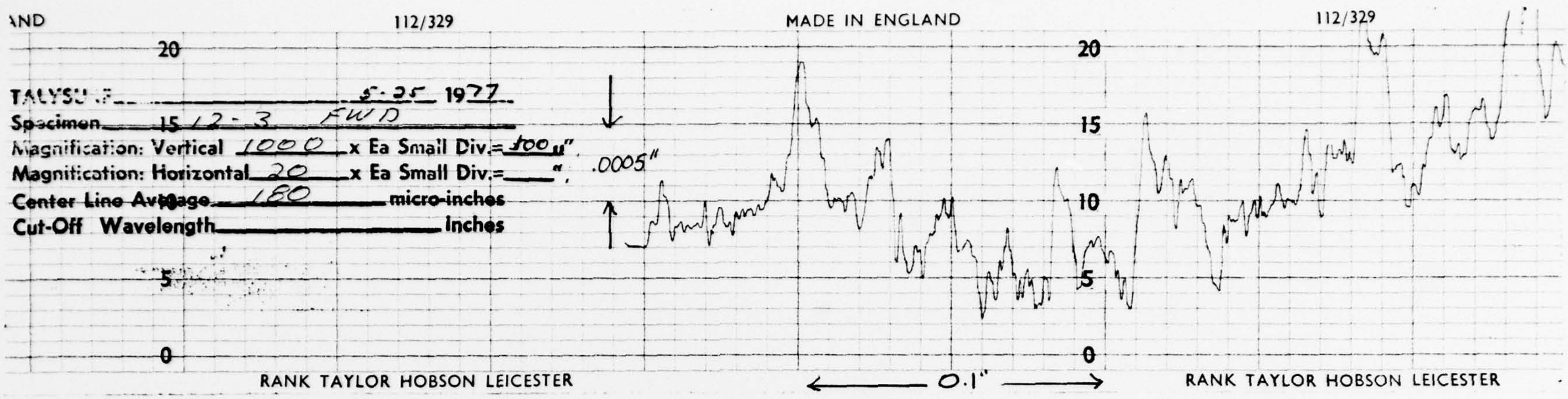


Fig. A4-11 Specimen 12-2. Exposed enamel (25 cals/cm^2) over 20 mil fiberglass skin.



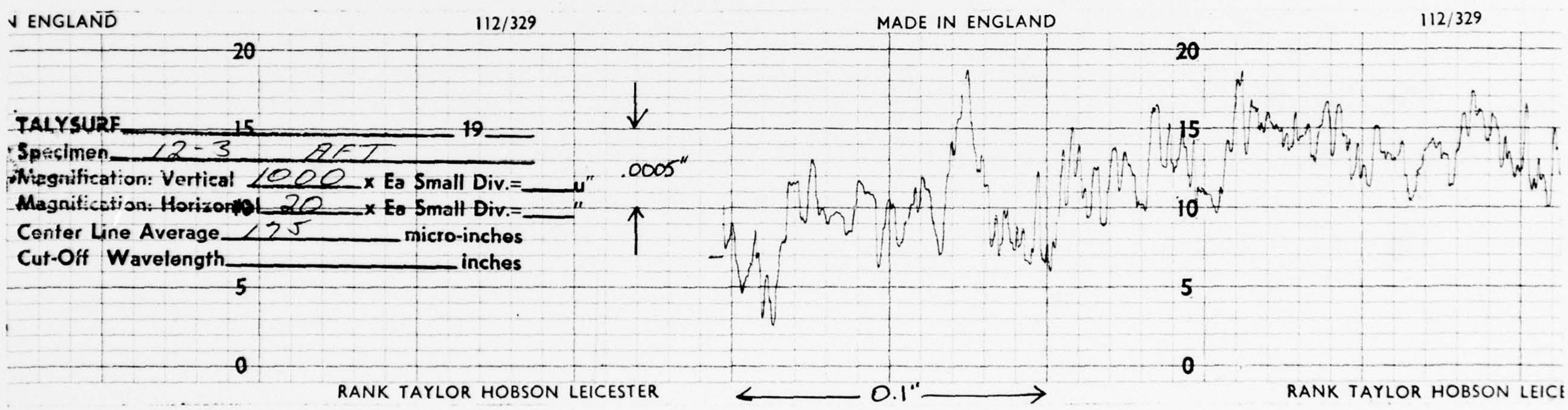


Fig. A4-12 Specimen 12-3 Fluence 38 cals/cm^2.

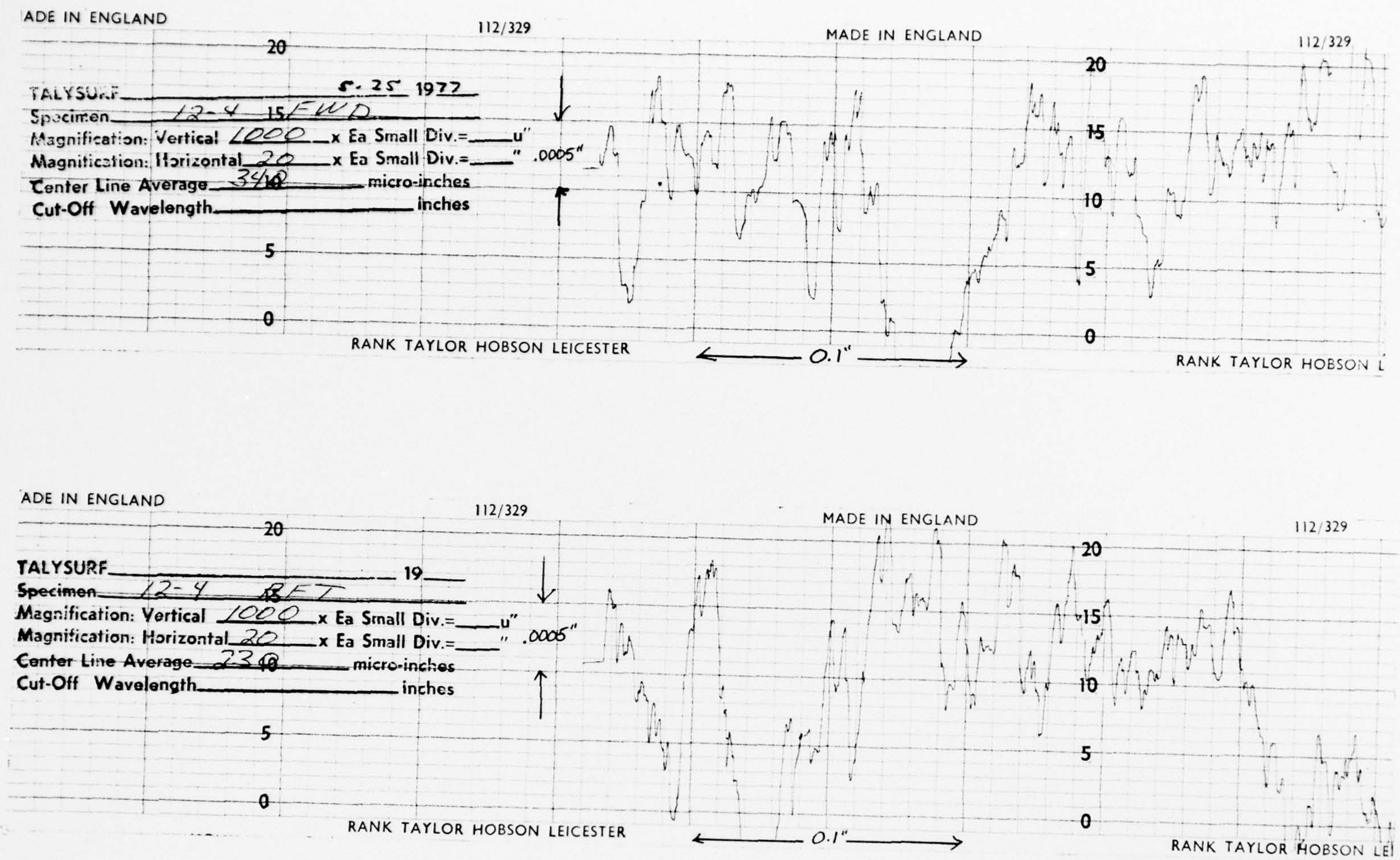
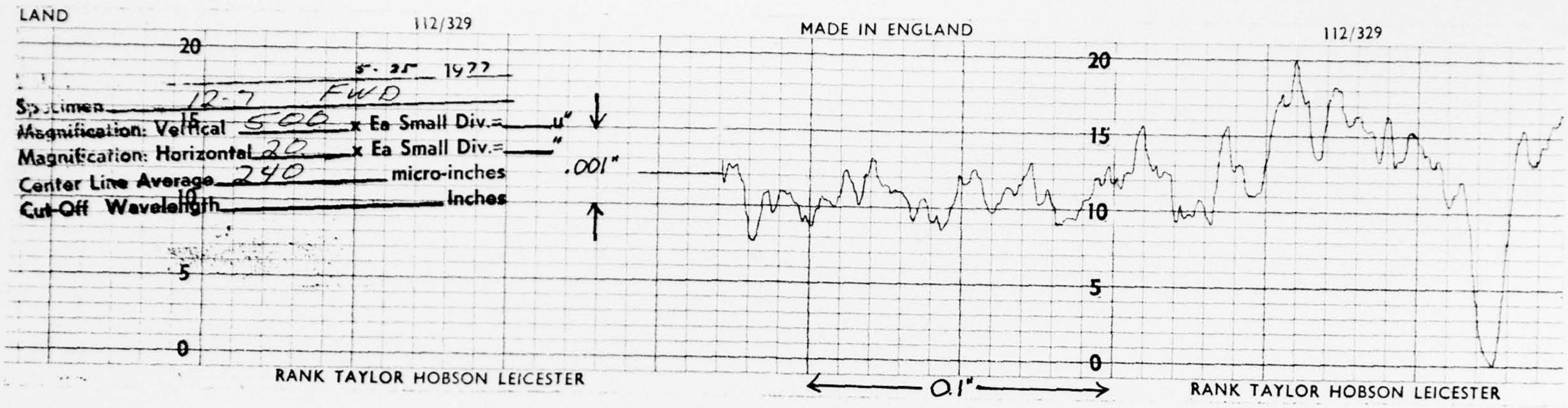


Fig. A4-13 Specimen 12-4. Fluence 64 cals/cm^2.



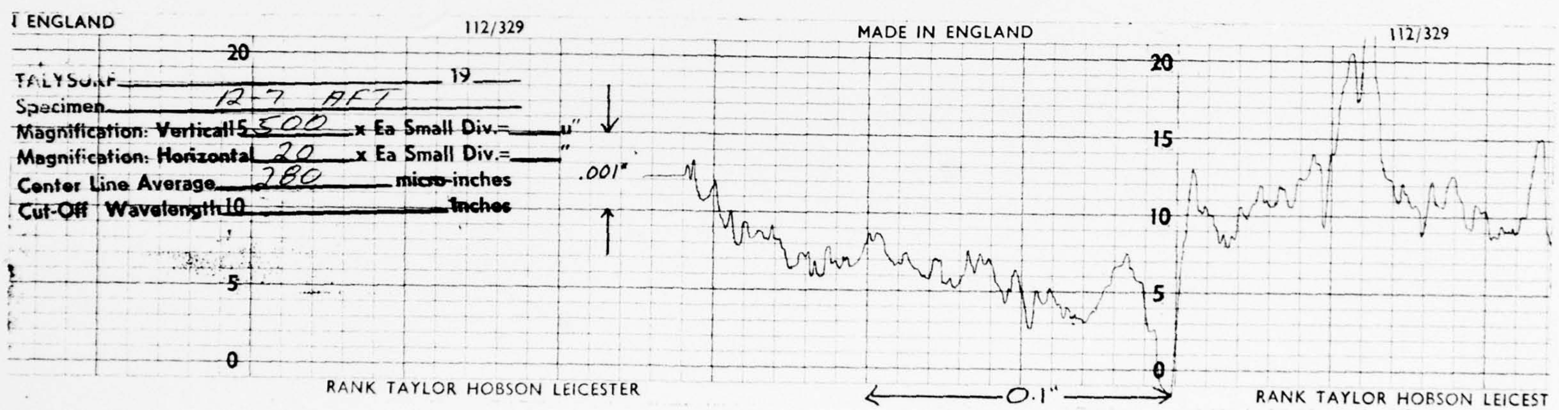


Fig. A4-14 Specimen 12-7. Fluence 82 cals/cm^2.

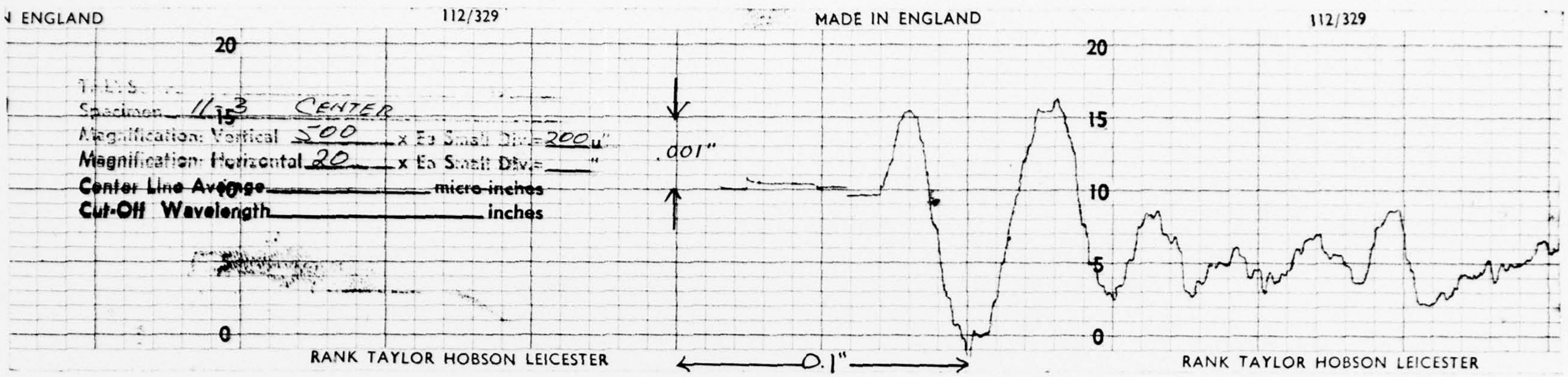


Fig. A4-15 (a) Specimen 11-3. Exposed enamel (25 cals/cm) over 13.5 mil fiberglass skin. Profile is in area of pinpoint blistering.

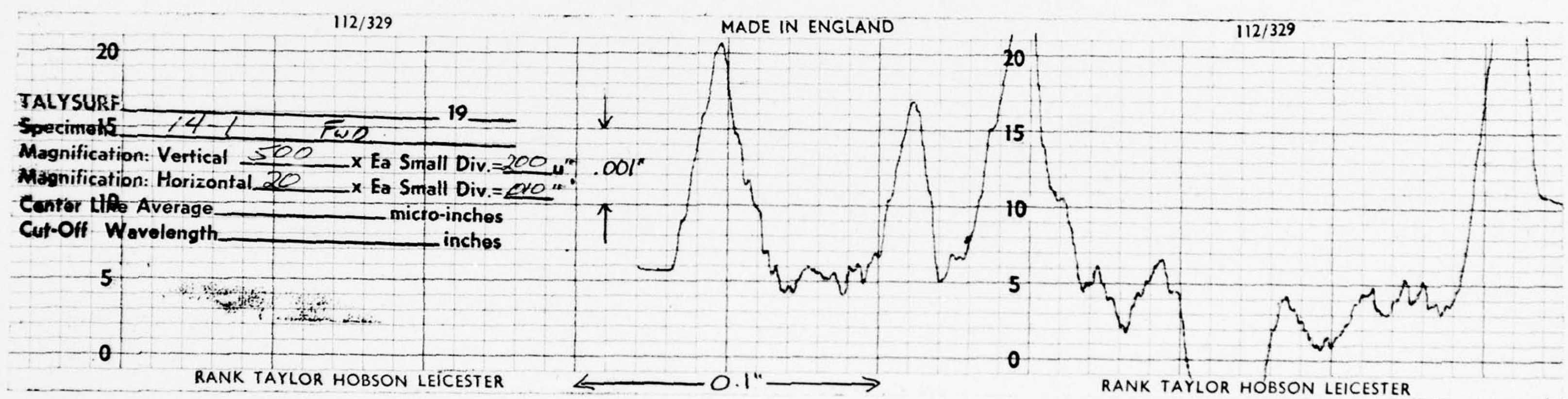
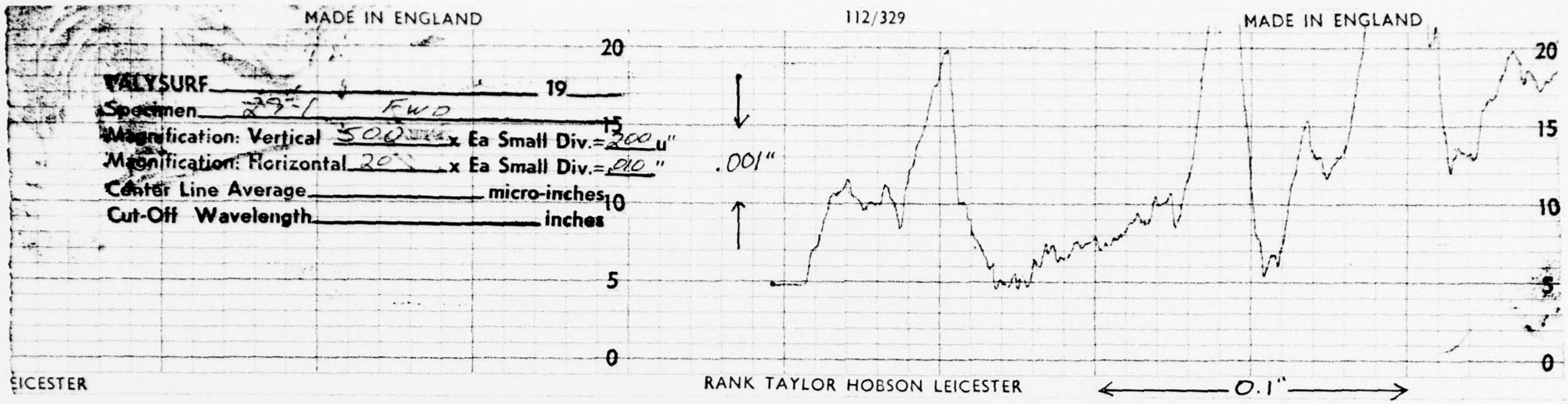


Fig. A4-15 (b) Specimen 14-1. Exposed enamel over 13.5 mil fiberglass skin. Profile is in area of pinpoint blistering.



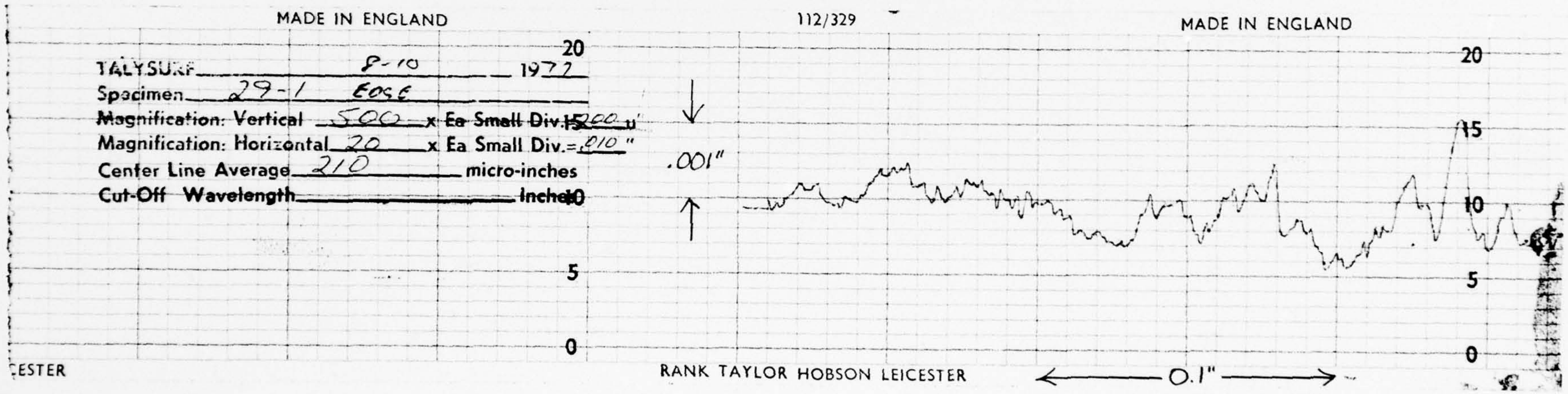


Fig. A4-16 Specimen 29-1. Exposed enamel (31 cals/cm^2) over 20 mil graphite skin. Upper profile is in area of pin-point blistering.

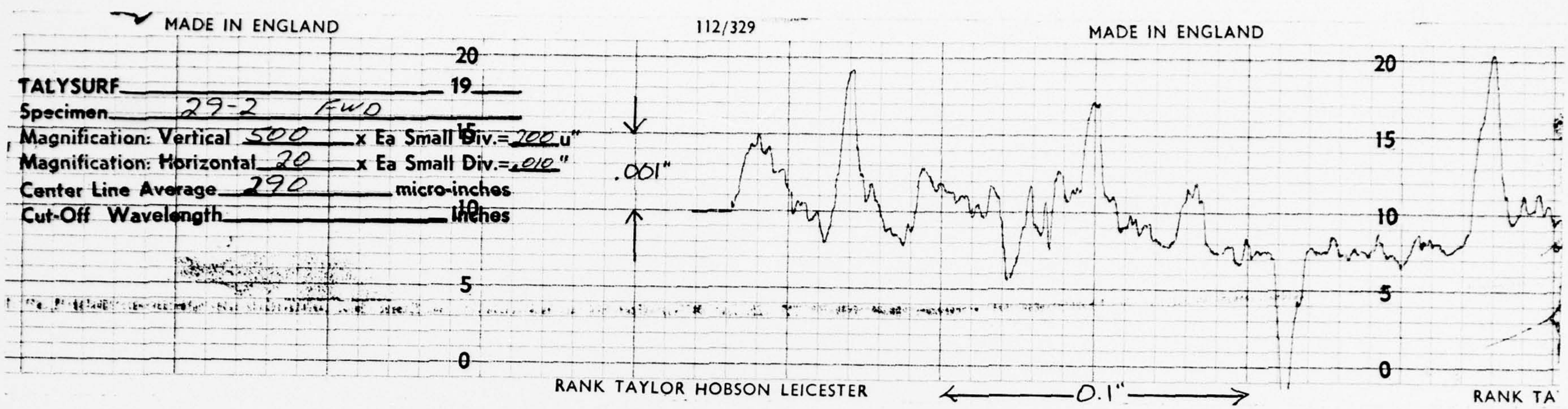


Fig. A4-17 Specimen 29-2. Fluence 49 cals/cm^2.

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